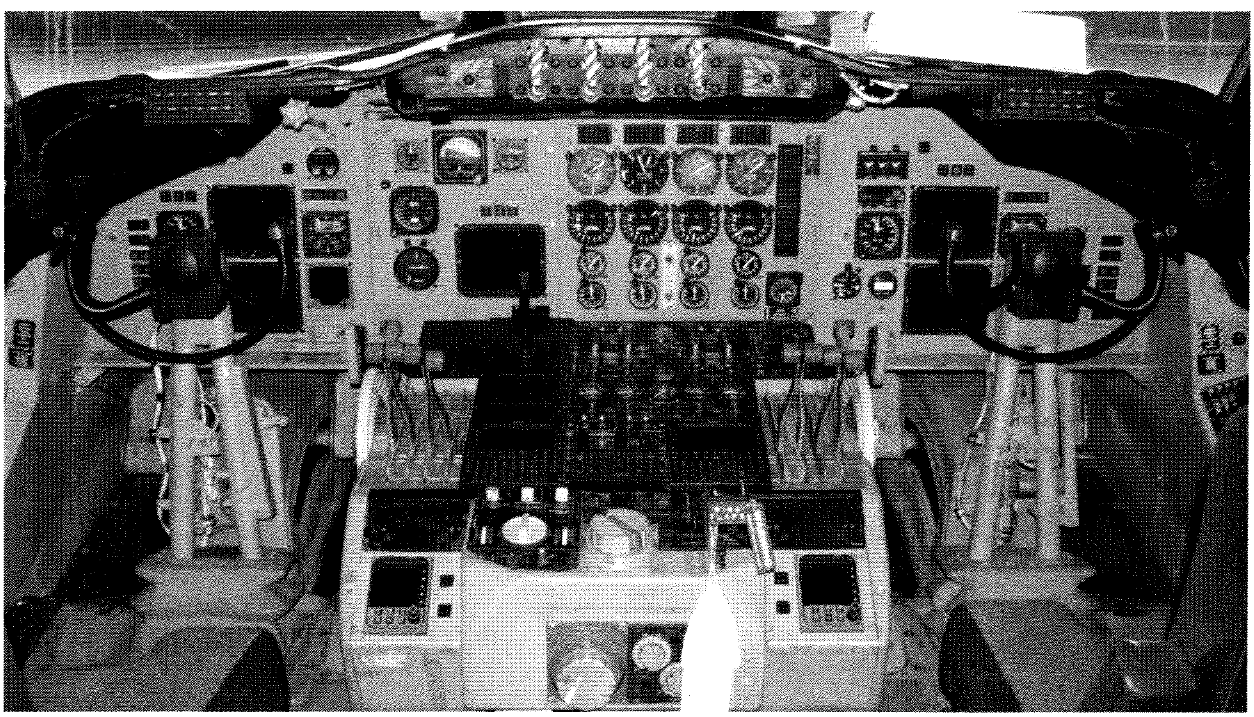
NASA P-3B Autopilot / Avionics Upgrade
Statement of Work
January 25, 2010

Introduction:

NASA Goddard Space Flight Center's Wallops Flight Facility (WFF) operates a P-3B Orion (BUNO 152735) aircraft designated as N426NA. The aircraft has been extensively modified to meet the needs of the NASA Airborne Science Program community. This Statement of Work details specifications for the design, analysis, acquisition, installation, testing, and acceptance of an autopilot and avionics upgrade. This upgrade package shall integrate with existing avionics and radar systems and include the following:

- Digital autopilot and flight director system
- Flight management computer system
- Digital pilot displays
- Digital center map display
- Digital engine instrumentation displays
- TCAS - Traffic /Alert and Collision Avoidance System
- TAWS - Terrain Awareness and Warning System

The specific work requirements are described in detail in section 6 of this document and Appendix A, Detailed Digital Autopilot Specification.



Current N426NA Cockpit as of 6/2008

Note:
N426NA has had the US Navy P-3 navigation interconnection box (Nav-J box) removed. Industry standard ARINC 429 avionics are currently installed. Applicable existing equipment is outlined in Appendix B and Appendix C. Vendors shall plan to use available rack space in the existing forward and aft avionics bays. Rack space may be added with NASA approval.

1.0 General Requirements:

1.1 Purpose

The vendor shall perform all necessary work (design, analysis, acquisition, manufacture, installation, ground testing, and training) to install the above listed upgrade package on N426NA.

After contract award, NASA shall provide full access to the aircraft for the agreed installation period. The contracted vendor is able to make trips to the P-3 during the design and manufacturing phases for design verification and fit check purposes. Access request shall be made one week in advance to the project manager. The vendor may request additional aircraft availability for Wallops Flight Facility site visits. Such availability is contingent upon operational and maintenance requirements and is at the discretion of NASA.

Final installation, ground testing, and training shall occur at the vendor's facility. The final product shall comply with all applicable Federal Aviation Agency (FAA) instrumentation standards and be capable of FAA instrument flight certification for enroute, departure, arrival, and terminal procedures therein. The delivered system shall interface with existing Inertial Navigation Systems (INSs), Air Data Computers (ADCs), transponders, navigational radios, communications radios, and standby instrumentation in order to minimize overall system cost.

This project is part of the American Recovery and Reinvestment Act (Recovery Act) of 2009 (ARRA).

1.2 Period of Performance

The period of performance for this contract shall be from date of award to August 31, 2010.

Physical installation of equipment to the aircraft to include testing shall be:
July 1, 2010 to August 31, 2010.

All design, analysis, physical aircraft modifications, tests and deliverables shall be received / completed and accepted by NASA no later than: August 31, 2010.

1.3 Purchase/Installation Options

The purchase and installation shall consist of the following:

Base Bid - Digital Autopilot System (DAS) and Flight Director System (FDS)

In addition to the Base Bid, the Government may elect to include any or all of the following options:

Option #1 - Two flight management computers (FMCs) and flight display interface. Two sets of dual digital pilot displays

Option #2 - Digital center map display system with TAWS

Option #3 - Digital engine instrumentation displays

1.4 Aircraft Systems Use

Upon delivery to the contract vendor's facility, the vendor shall not operate any P-3 aircraft system not directly related to the execution of this contract. The installation shall be completed using external electrical power sources meeting the specifications outlined in the original equipment manufacturers operations manual (NATOPS/01-75PAC-1). If use of the Auxiliary Power Unit (APU) is required for short durations, the onsite NASA representative shall be present. NASA shall assign a project manager to act as the single point of contact for day-to-day interaction as well as the monitor of the overall project timeline. The contract vendor shall only contact the NASA Contracting Officer's Technical Representative (COTR) or alternate COTR with questions and requests. NASA shall provide a response no later than 5 business days from initial contact concerning a question or request. The NASA COTR and alternate COTR for this installation shall be:

Contracting Officer's Technical Representative (COTR):
Ed Sudendorf – NASA Aircraft Office
NASA Wallops Flight Facility
Bldg. N-159 Hanger
Wallops Island, VA 23337
Phone: 757-824-1240
Fax: 757-824-2135
Cell: 757-894-3753

Alternate Contracting Officer's Technical Representative (COTR):
Anthony Guillory – NASA Aircraft Office
NASA Wallops Flight Facility
Bldg. N-159 Hanger
Wallops Island, VA 23337

Phone: 757-824-2161
Fax: 757-824-2135
Cell: 757-894-5967

1.5 Starting the engines, flight or taxi of the aircraft under its own power is prohibited.

1.6 Current Documentation Provisions
NASA shall also provide the contracted vendor with full access to any applicable and available N426NA structural manual, engineering analysis, electrical diagrams, or existing avionics documentation. Certain documents shall only be available as scanned electronic copies.

2.0 Vendor Requirements:

2.1 Vendor Personnel
The vendor shall provide technicians and/or mechanics who hold a valid FAA - Airframe and Powerplant License and have previous experience installing such equipment to a P-3B Orion.

2.2 Vendor Maintenance Facility
The vendor shall perform all aircraft modifications and installation at their designated facility.

The reposition of N426NA is the responsibility of NASA.

The facilities which house NASA 426 during the entirety of this contract shall be an FAA part 145 or DCMA certified repair station and provide (at a minimum) the following:

i. Total aircraft enclosure from the elements throughout the duration of the contract
ii. Aircraft fire detection and suppression systems (must be integral to the hangar)
iii. Aircraft firefighting and rescue capabilities available at the facility must adhere to NASA-STD- 8719.11. See NFPA 403 and CFR 14 Part 139 for aircraft firefighting requirements of NFPA Category 6 (FAA Category B).
iv. 24 hrs aircraft security with limited access via guarded entryways to the hangar facility and a maximum of one security personnel (may be supplied by the airfield)

2.3 PreAward Site Visit with Aircraft Availability

In order to provide the most competitive cost estimate, all offerors are required to make one site visit to inspect the aircraft prior to submitting a proposal.

2.4 Changes to Existing Aircraft Components or Structure

If during the installation process, the vendor indentifies deficiencies in, or required upgrades to, existing components or structure that may be necessary to interface with the new avionics and autopilot systems, the vendor shall contact NASA immediately.

If existing hardware needs to be replaced, modified or repaired the vendor shall notify NASA. NASA shall provide this hardware or designate the vendor to perform such repairs, modifications or replacement. NASA may elect to extend the delivery date of this contract by the time required for such action. However, if the indentified hardware deficiency, modification, or required upgrade, is deemed by NASA to have been reasonably identifiable by the vendor during the planning and bidding phase of this contract, NASA may hold the vendor responsible for all, or part, of the cost and time for such replacement, modification, or upgrade.

All removed or replaced components, or subsystems thereof, remain the sole property of NASA. The relocation of any existing / remaining instrumentation shall be approved by NASA.

3.0 NASA Responsibility:

NASA personnel or personnel from independent contractors (to be determined) shall be allowed to monitor the aircraft throughout the installation process. In accordance with NASA Procedural Requirement (NPR) 7900.3B, NASA representative maintenance personnel shall act as NASA's governmental quality assurance representatives throughout the installation process. This process is on a not-to-interfere basis and shall not affect the overall timeline.

4.0 Facility, Maintenance and Quality Assurance:

4.1 Overall Quality

All analysis and installations shall meet NASA specific standards as outlined in Appendix D (548-RQMT-0001, P-3 Design Requirements).

Upon contract completion, the aircraft shall comply with all applicable FAA instrumentation standards and be capable of FAA RVSM (Reduced Vertical Separation Minima), as well as, instrument flight certification for enroute, departure, arrival, and terminal procedures therein.

Offerors shall include a Quality Assurance Plan with the proposal that details the offeror's plan of compliance with the NASA/FAA standards identified above. This plan shall be reviewed and approved by NASA prior to the commencement of this contract and incorporated as part of Section J, Attachment B of the contract.

4.2 Tool Control Program

In accordance with NPR 7900.3B sections regarding aircraft maintenance activities, the vendor shall maintain a tool control program.

4.3 Foreign Object Damage (FOD) Control Program

In accordance with NPR 7900.3B sections regarding aircraft maintenance activities, the vendor shall maintain a FOD control program.

5.0 Engineering and Review Deliverables:

5.1 Project Status Tracking

The vendor shall provide monthly progress reports detailing current status and progress toward schedule milestones. These reports shall be provided electronically (i.e. email) to the assigned NASA Project Manager and the Contracting Officer's Technical Representative (COTR).

5.2 Mechanical Engineering Analysis

All mechanical engineering analysis shall show positive Safety Margins with current NASA P-3 standards: crash loads, gust loads, aerodynamic loads (if applicable) and Factors of Safety values for the intended installation location.

5.3 Electrical Engineering Analysis

All electrical circuits shall be protected in accordance with FAA industry aviation electrical standards.

Additionally, pilot and copilot primary flight instrumentation, navigation, and communication equipment shall be powered by mutually exclusive power, and /or control, circuits. Any single electrical bus failure shall not eliminate the aircraft's ability to: externally and internally communicate, navigate, or cause the complete loss of primary flight instrumentation.

It is required that the pilot-side (left seat position) primary flight instruments (including: (turn needle, airspeed, altitude, heading, attitude indication, flight management computer, one VHF radio, one VHF Omnidirectional Range (VOR)/ (Instrument Landing System) ILS navigation radio-capable of a precision approach, and flight director system) be powered by either the Monitorable

Essential AC bus or the Flight Essential AC bus (where AC power is required) and the Monitorable Essential DC bus or Flight Essential DC bus (where DC power in required). The purpose of this requirement is to preclude P-3 'Load Monitoring' as described in the P-3 Flight Manual (NATOPS/01-75PAC-1) section 2.2.5, from affecting safe aircraft operation, navigation, and communication, under instrument meteorological conditions.

5.4 Thermal Engineering Analysis

Thermal design standards shall provide for continuous component operation temperatures at least 10% below manufacturer suggested maximum operating temperatures. Under all circumstances, avionics equipment must be able to sustain continuous operation with a supplied cabin static air temperature range of 1°C - 29°C.

5.5 Data Format and Ownership

All hardware, drawings and analysis (paper and electronic) and all other documentation generated as a result of the execution of this contract shall be the property of NASA upon the successful completion and acceptance of the avionics upgrade. All technical data shall be provided in electronic format on CD or DVD.

5.6 Preliminary Design Review (PDR) data package:

The vendor shall present their design at a NASA Preliminary Design Review (PDR) and Critical Design Review (CDR) and shall receive NASA approval prior to the start of any manufacturing, fabrication or installation. Within the scope of the original contract, the vendor shall comply with all action items assigned through the NASA review process. The vendor shall participate via teleconference for each of the reviews. The review should be no longer than 3 hours in duration and provided in writing/presentation and verbally.

NASA shall review and approve all thermal, mechanical, and electrical analysis and installation instructions, including drawings, prior to aircraft modification. All thermal, mechanical and electrical drawings and models shall be supplied electronically. The formats are Autodesk AutoCAD 2009, Autodesk Inventor 2009 or earlier version of each of the specified programs. Upon acceptance, these items become the sole property of NASA.

The PDR date shall be completed within thirty (30) working days of the contractual start date. The PDR package is due three (3) working days prior to the scheduled review. The PDR data package shall include the most up-to-date version of the following:

- Preliminary models and/or drawings showing P-3 modifications and flight station layout
- A list of structural, thermal and modal (if applicable) loads / values to be used for final analysis

- Preliminary structural analysis showing feasibility of modification
- Preliminary electrical schematics and load analysis
- Preliminary signal flow diagrams

5.7 Critical Design Review (CDR) data package:

The CDR date shall be completed within twenty (20) working days of the completion of PDR. The CDR package is due three (3) working days prior to the scheduled review. The CDR data package shall include the most up-to-date version of the following:

- Finalized model, manufacturing and installation drawings of P-3 modifications
- Finalized engineering reports:
    - Mechanical
    - Electrical
    - Thermal
    - Modal (if required)
- Finalized electrical schematics and load analysis
- Finalized signal flow diagrams
- Preliminary training plan for Maintenance and Flight Crew personnel
- Preliminary System Acceptance Test Profile (SATP)

5.8 The NASA Airworthiness Process

NASA is required to perform a Configuration Review (CR) and Final Installation and Inspection Review (FIIR) as part of the NASA Airworthiness Process. These reviews are internal to NASA however; the contracted vendor is required to attend via teleconference in both airworthiness reviews as action items from these reviews may be forwarded to the vendor for compliance within the scope of the original contract.

5.9 Avionics Accessibility

The installation design shall allow for access to avionics instrumentation, both new and existing, in the avionics bays and flight station for replacement and / or repair.

5.10 Final Data package:

The finalized data package shall be submitted to NASA after the SATP (System Acceptance Test Profile) acceptance and signifies the completion of this contract. This data package includes the following items in their final configuration:

- Complete set of manufacturing and installation drawings and models, as well as installation instructions and maintenance procedures (as required and outline in the above statement of work). All drawings and models shall be supplied electronically and in a format that can be opened with Autodesk AutoCAD 2009 or Autodesk Inventor 2009 or earlier version of the specified programs where applicable.

- Copies of all installation related analysis (thermal, structural, electrical, modal - if required, etc) - Native files of all drawings and analysis used for this installation.
- Data package shall be prepared in contractor's chosen format however; technical review submittals must be in PDF format.

6.0 Vendor Specific Hardware Deliverables:

6.1 Flight Hardware

All hardware and materials used shall be aircraft quality with component certificates provided to NASA, as applicable. All flight systems shall be in compliance with standards set forth in FAA Order 8900.1 Flight Standards Information Management System (FSIMS). The system must be capable of international operations. Unless otherwise noted, component airworthiness and standards are set forth in Federal Aviation Regulation Part 25 – Airworthiness Standards: Transport Category Airplanes.

6.1.1 Digital Autopilot System (DAS) and Flight Director System (FDS) – The DAS shall provide fail-safe passive redundancy in all three axes. In the event that data from one of the Inertial Reference Units (IRUs) becomes inoperative, single IRU operations shall be possible. The DAS functions shall include separate pitch and roll functional modes including: attitude hold, heading/track hold, control wheel steering, barometric altitude hold, radar altitude hold, barometric altitude selection & capture, indicated airspeed hold, and vertical speed hold. It shall also allow the pitch and roll modes to be selected independently.

The DAS and FDS shall comply with the Detailed Digital Autopilot Specification, Appendix A.

The DAS shall incorporate: yaw damping, automatic turn coordination, and automated electric elevator trim.

The DAS shall be capable of FAA RVSM (Reduced Vertical Separation Minima) certification under Federal Aviation Regulation Part 91 Appendix G.

The Flight Director System shall be capable of providing pilot aircraft attitude cuing.

6.1.2 Two flight management computers (FMCs) and flight display interface– Dual FMCs shall allow the aircrew to use the Global Positioning System (GPS) as the primary means of navigation for

Required Navigation Performance (RNP) Area Navigation (RNAV) operations during all phases of flight, including approach. The FMC shall allow the operator to enter data via a keypad / scratchpad entry system (or similar) to perform required navigational calculations, present textual display of data, and provide steering guidance to the MFDs. The system shall provide positioning data and GPS system integrity monitoring functionality. Steering guidance and a pictorial representation of the flight path including: airports, VORs, Tactical Air Navigation (TACAN)s, combined VOR and TACAN (VORTAC)s, Non Directional Beacons (NDBs), intersections, approach-arrival-departure waypoints and other information that may enhance situational awareness. Applicable alerts (such as system degradation or unreliability) shall be displayed. The system should be capable of declutter mode on the MFDs in order to minimize screen clutter. The navigation database stored in the FMC shall be easily updatable and flight plans shall be programmable from an external system and imported via easily obtainable and commercially available portable media devices.

The FMCs shall be capable of producing, and providing steering to, an internally generated approach path for non-precision approaches where external precision approach and glideslope information is unavailable.

The FMCs shall be capable of operations at all latitudes, including both Polar Regions.

6.1.3 Two sets of dual digital pilot displays – The Electronic Flight Displays shall be used to provide heading, track, course, and attitude information to the pilot and copilot via identical, interchangeable Multi-Function Displays (MFDs) with reversionary failure modes. Each unit shall use color and symbol shapes to distinguish significant or abnormal situations from normal display functions. The MFD shall continuously monitor navigation parameters and create alerts when an abnormal flight situation is detected. The alerts shall be displayed as flags, highlights and / or removal of related navigation data. The MFDs shall use standard International Civil Aeronautical Organization (ICAO) symbology and associated identifiers in the generation of standard navigational and flight data. Additional display information available to the pilot and copilot MFDs shall consists of the following:

1. Heading bugs and ground track display.
2. Bearing and heading source data.
3. A graphical depiction of the great circle flight plan route displaying the aircraft position relative to the waypoints.
4. Marker beacon information.
5. Glideslope (or FMC generated precision path) and localizer scales and indicators.

6. A turn needle depiction based on computed turn rates.
7. Selectable real time wind readout.
8. FMS waypoint distance.
9. VOR, TACAN, Distance Measuring Equipment (DME), NDB information.
10. Receiver Autonomous Integrity Monitoring (RAIM) annunciations.
11. Selectable background data (airports, Navigational Aids, intersections).
12. Origin and destination airports.
13. FMS vertical navigation (VNAV) path advisory information.
14. Selectable display of progress information.
15. Selectable display of secondary flight plan.
16. Display of modified flight plan and offset route.
17. Message alerts
18. Color weather radar depiction (if not depicted on center panel)
19. Map mode display (if not depicted on center panel)
20. Visual terrain depictions TAWS (if not depicted on center panel)
21. TCAS information (if not depicted on another system)

6.1.4 Digital center map display system– The center map display shall be capable of displaying a moving map with selectable alternate display modes. It shall also be capable of displaying a third party digital data input (FALOCNVIEW or similar software driven from an external non-included system).

6.1.5 Digital engine instrumentation displays – The digital engine instrumentation displays shall provide for centralized engine instrumentation and annunciation and allow for precise monitoring of engine system condition.

7.0 Training:

All training shall occur after the installation is completed and prior to flight test of the installation.

7.1 Personnel Training

The vendor shall provide NASA flight crew and maintenance personnel (total number not to exceed: 4 maintenance training personnel and 4 flight crew training personnel) initial operational and maintenance training of all upgraded equipment. The instructors, duration, and content of the training shall be selected by the vendor and performed at the vendor's facility in a manner so as to provide overall system understanding.

7.2 Maintenance Training

This maintenance training shall be accomplished at a vendor selected location and encompass Operational Level (O-Level) daily maintenance of the systems and software to include, but not limited to: systems overview, system fault recognition, repair and replacement, and preventative maintenance procedures.

7.3 Flight Crew Training

This flight crew training shall be accomplished at a vendor selected location and encompass daily operational use of the systems and software to include, but not limited to: systems overview, system fault recognition, default modes of operation, flight display modes, flight plan insertion, flight plan navigation, symbology, coupled and uncoupled approaches, departures and arrivals.

8.0 NASA Acceptance of Vendor Deliverables:

8.1 Operating Manuals and Documentation

The vendor shall provide all applicable operating manuals, documentation, licensing materials, updating websites, and required operational / updating software pertaining to the avionics upgrade to NASA. At the time of acceptance, all systems shall be installed with the latest approved software versions, revisions and databases. The vendor shall provide NASA with all required information to update all avionics upgrade system and support software and databases.

8.2 Installed Equipment

The vendor shall provide the name, manufacturer, and part-number of each upgraded component and its supplier such that NASA may acquire components external to this contract.

8.3 Test and Repair Equipment

The vendor shall provide all specialized test and repair equipment (if any) needed for operational level maintenance of all newly installed or modified components.

8.4 Hardware Warranty

All installed equipment, hardware, software, mounting structure, and support equipment shall be warranted under original manufacturer's warranty to best extent possible and shall prove to be free from manufacturing and /or installation defect prior to the time of aircraft acceptance by NASA.

9.0 Contract Completion / NASA Acceptance:

9.1 Test and Evaluation / Aircraft Acceptance

The vendor shall develop and present to NASA for approval and acceptance a 'System Acceptance Test Plan' (SATP) to adequately flight test the avionics upgrade installation. The profile shall be flown by NASA aircrew. Vendor aircrew may accompany the SATP flight as approved by the NASA GSFC/WFF Aircraft Office. Acceptance of the system as operationally meeting this SATP shall be at the discretion of the NASA pilot flying the SATP.

Overall system functionality and ease-of-use will be evaluated by the NASA Test Crew using the Bedford Workload Scale for the given task and flight phase identified in the SATP. The Bedford Workload Scale is a well-established, and the most widely accepted, scale for assessing pilot workload while performing a prescribed task. It is measured in a value ranging from 1 to 10 and denoted as Handling Qualities Rating (HQR).

Deficiencies requiring attention, if any, will be identified by the following Bedford workload rating values:

- Level 1 **Satisfactory**
  - Workload is clearly adequate for task & flight phase
  - Equate to Bedford HQR 1-3
- Level 2 **Acceptable**
  - Workload is adequate to accomplish task during the prescribed flight phase however, an increase in pilot workload exists
  - Equate to Bedford HQR 4-6
- Level 3 **Unacceptable**
  - Qualities such that pilot workload is excessive for task during the prescribed flight phase
  - Equate to Bedford HQR 7-9+
  - Not necessarily defined as safe

Level 1 Deficiencies will require no action on the part of the contractor.
Level 2 Deficiencies will be recommended for corrective action to the contractor and may be conducted at the contractor's discretion or agreed to be completed at a future time.
Level 3 Deficiencies will not be accepted. Corrective action is required by the contractor. The SATP will not be considered complete until all Level 3 deficiencies are corrected or eliminated as evaluated by the assigned NASA pilot.

9.2 Final Acceptance

Final acceptance of all deliverables shall be upon the successful completion of the SATP by NASA and the delivery of the final data package at the vendor's selected location.

Detailed Digital Autopilot Specification

1.0 General System Overview & Scope

The Digital Autopilot System (DAS) shall replace the functionality of the existing PB-20N Autopilot system with a digital, highly reliable, fail-safe system, add new operational modes

The P-3 DAS shall be a digital replacement for the existing PB-20N Autopilot system. The DAS shall provide all existing Autopilot functions and provide enhanced modes of operation.

The nomenclature, designation, or terminology may vary however, the DAS modes must functionally include (at a minimum) Pitch and Roll Attitude Hold, Heading Hold, Barometric Altitude Hold, Radar Altitude Hold, Pre-select Heading, Pre-select Altitude Capture, Navigation (NAV) mode, Approach mode, Control Wheel Steering, Automatic Pitch Trim, pitch, roll, and yaw rate damping, and Indicated Air Speed (IAS) Hold.

The DAS shall be a dual redundant system providing full fail-safe / fail-soft operation.

The DAS shall provide control for the existing P-3 aircraft flight control surface actuation devices, including hydraulic booster assemblies, Autopilot Engagement Valve, Hydraulic Load Sensors, Hydraulic Load Control Unit, Hydraulic Transfer Valves, and Elevator Trim Servomotor.

The DAS must interface with the existing Flight Management Computer (FMC) or an FMC integrated with this upgrade.

The DAS shall also provide ARINC 429 digital bus interface for supporting navigation (NAV) modes.

The DAS shall provide a Built-In Test (BIT) and monitoring to continually assess system availability and provide fail-safe operation.

Monitoring shall also be provided for each hydraulic booster assembly to detect actuator component failures. The DAS shall also provide preflight and ground test capability for the Autopilot system. During flight, the DAS shall provide failure and deviation warnings to a Caution and Warning system.

2.0 Applicable Documents

The following documents, current edition, or any previous edition up to the specified date, form a part of this specification to the extent specified herein. In the event of conflict between the documents referenced herein, the NASA Statement of Work, and the contents of this specification, the contents of this specification shall be considered as superseding requirements.

Referenced Documents

MIL-HDBK-5400 30 November 1995	Electronic Equipment, Aircraft, General
MIL-STD-464 18 Mar 1997	Electromagnetic Environmental Effects Requirements for Systems

Referenced Manuals/handbooks

MIL-HDBK-217F 28 Feb 1995	Reliability Prediction of Electronic Equipment
MIL-HDBK-5J 31 Jan 2003	Metallic Materials and Elements for Aerospace Vehicle Structures

Referenced Standards

MIL-STD-882D 10 February 2000	System Safety Program Requirements
MIL-STD-1472F 23 Aug 1999	Human Engineering Design Criteria for Military Systems, Equipment and Facilities
MIL-STD-704F 12 Mar 2004	Aircraft Electric Power Characteristics
MIL-STD-810F Notice 1 1 November 2000	Environmental Test Methods
NAVAIR 01-1A-505 01 Jun 2006	Installation Practices, Aircraft Electric and Electronic Wiring
MIL-F-9490 22 Apr 2008	Flight Control Systems – Design, Installation, and Test of Piloted Aircraft
MIL-W-5088L 5 Oct 1991	Wiring, aerospace vehicle
MIL-DTL-81381B 8 Sep 1998	Wire, electric, polyamide-insulated, copper or alloy

NPR 8705.2	NASA Procedural Guideline
NASA SOW	NASA P-3B Autopilot / Avionics Upgrade Statement of Work

3.0 Design requirements

The P-3 Digital Autopilot System (DAS) shall conform to the requirements of this specification. The DAS shall meet these requirements during both ground and airborne operation defined in this specification and in the applicable referenced documents.

The replacement Autopilot shall be integrated with the existing systems on the P-3 Orion aircraft and be capable of future expansion. With exception to the PB-20N Autopilot system, current display systems, and flight management computers, NASA wishes to minimize changes to the existing systems fitted to the P-3 Orion aircraft.

4.0 Operational requirements

The DAS shall provide the capability to automatically control the flight control surfaces of the P-3 aircraft for all operational roles that require automatic control. The DAS shall operate as specified herein, but may exhibit reasonable reduced parameter accuracy while the configuration of the aircraft is transitioning from one state to another (i.e., Altitude Hold Error during flap and/or landing gear extension). The DAS shall maintain aerodynamic balance in pitch, roll, and yaw during aircraft weight and balance changes, and aircraft power and speed changes, throughout the aircraft operating envelope.

The DAS shall meet or exceed the performance characteristics specified herein.

4.1 Basic criteria

The DAS shall provide the following essential capabilities:
a. Set and hold aircraft attitude in pitch and roll;
b. Set and hold the existing heading;
c. Set and hold aircraft barometric altitude;
d. Set and hold radar altitude; (within radar altimeter range)
e. Set, fly to, and hold a pre-selected heading or course;
f. Set, fly to, and hold a pre-selected barometric altitude;
g. Visually and audibly indicate to the flight deck aircrew the pre-selected altitude is being obtained;
h. Provide visual and audio cautions and warnings related to DAS operation;
i. Provide Control Wheel Steering (CWS) to enable the flying pilot to steer the aircraft while maintaining barometric or radar altitude;

j. Provide CWS to enable the flying pilot to maneuver the aircraft in pitch and roll to a new desired attitude without disengaging the Autopilot;
k. Maintain automatic elevator trim during Autopilot operation;
l. Provide pitch, roll, and yaw rate damping;
m. Maintain the aircraft's aerodynamic balance (in pitch, roll, and yaw) by compensating for changes in aircraft power and speed throughout the P-3 Orion flight envelope, including three-engine asymmetric powered flight;
n. Perform Autopilot functions in all aircraft configurations including flap, landing gear, external configurations consistent with P-3 standard external stores configurations, and aircraft weight and balance changes including weight changes due to fuel burn/dump and internal/external stores release.
o. Ability to be used in conjunction with a flight director system and flight management computer
p. Ability to fly pre-programmed flight plans using the FMC
q. Ability to perform coupled Cat I Instrument Landing System (ILS) approaches using raw data
r. Ability to perform non-precision approaches using the FMC

4.2 DAS performance requirements
The DAS shall be capable of being engaged, operated, and disengaged throughout the P-3 Orion flight envelope, during all types of operations requiring automatic control, as defined in the P-3 Orion Flight Manual and In Accordance With (IAW) the P-3 Operational limitations.

The DAS shall maintain aerodynamic balance in pitch, roll, and yaw during aircraft weight and balance changes, and aircraft power and speed changes, throughout the aircraft operating envelope. Autopilot parameters are detailed below.

4.2.1 Heading Hold
When the DAS is engaged with a bank angle equal to or less than 2 degrees, and no other roll mode is engaged, the aircraft shall roll out wings level and hold the heading at the time of engagement within ±1 degree.

The DAS shall maintain the Heading Hold Mode until another roll mode is engaged. During heading hold engagement, the DAS bank angle limit shall be < ±20 ±2.0 degrees and the roll rate limit shall be ±15 ±1.5 degrees per second.

4.2.2 Roll Attitude Hold
The DAS shall provide a Roll Attitude Hold Mode. The Roll Attitude Hold Mode shall be engaged when the DAS is engaged with a bank angle of greater than ±2 ±0.2 degrees, no other roll mode is engaged.

The DAS shall hold the bank angle existing at the time of engagement. If the bank angle is greater than ±45 ±4.5 degrees at the time of engagement, the DAS shall return and hold the aircraft to a bank angle of ±45 ±4.5 degrees in an expeditious and non-abrupt manner.

The Roll Attitude Hold Mode shall be disengaged by the engagement of another roll mode.

During Roll Attitude Hold engagement, the DAS roll rate limit shall be ±15 ±1.5 degrees per second. Roll Attitude Hold shall be capable of being engaged throughout a minimum range of ±70 degrees of bank.

4.2.3 Pitch Attitude Hold
The DAS shall provide a Pitch Attitude Hold Mode. The Pitch Attitude Hold Mode shall be engaged when the DAS is engaged with no other pitch mode engaged. The DAS shall hold the pitch angle existing at the time of engagement. If the pitch angle is greater than ±22 ±2.5 degrees at the time of engagement, the DAS shall return and hold the aircraft to a pitch angle of ±22 ±2.5 degrees in an expeditious and non-abrupt manner.

The Pitch Attitude Hold Mode shall be disengaged by the engagement of another pitch mode.

During Pitch Attitude Hold engagement, the DAS pitch angle limit shall be ±22 ±2.5 degrees and the pitch rate limit shall be ±13 ±1.3 degrees per second. Pitch Attitude Hold shall be capable of being engaged throughout a minimum range of ±45 degrees of pitch.

Pitch Attitude Hold shall be augmented by a roll attitude signal to provide up elevator compensation in a turn.

4.2.4 Roll Control Wheel Steering (CWS)
A roll control wheel (either using the existing aircraft control column with force override; or a new Roll CWS input device) shall produce an aircraft roll rate proportional to the force (existing control column) or displacement (new CWS input device) applied. If the existing control column is used, the roll rate versus force gradient shall be constant throughout the aircraft flight regime as long as the bank angle is less than ±45 degrees. The roll control wheel steering function shall allow the pilot to maneuver the aircraft with the control wheel to the desired roll attitude while pitch axis modes are engaged.

If a new CWS device (independent of the control column) is utilized, its inputs shall not be allowed to exceed aircraft bank angles ±45 ±4.5 degrees.

4.2.4.1 Control column roll force gradient requirements
Roll CWS via the installed control column may be used to command the aircraft to bank angles greater than ±45 ±4.5 degrees. The nominal control column force gradient shall be 2.125 ±0.3 degrees per second of roll rate per pound of roll CWS force that exceeds the ±2 ±0.2 pound dead zone. When Roll CWS is used to command the aircraft to a

bank angle greater than ±45 ±4.5 degrees, the roll rate vs force gradient shall be reduced, requiring an increasing force to obtain the bank angle in excess of ±45 ±4.5 degrees.

When the roll CWS force is released with the bank angle greater than ±45 ±4.5 degrees, the autopilot shall return the aircraft to a bank angle of ±45 ±4.5 degrees. Maneuvers initiated by the use of roll CWS shall be coordinated without the necessity of manual control in pitch and yaw. During Roll CWS, the DAS roll rate maximum limit shall be ±30 ±3 degrees per second.

4.2.4.2 Control column pitch force gradient requirements
If the existing control column is used for pitch CWS inputs, control laws shall produce an aircraft pitch acceleration proportional to the force applied. The steady state control wheel force per "g" shall be constant when the aircraft pitch angle is less than ±22 ±2.2 degrees. If the existing control column is used to command the aircraft to a pitch angle greater than ±22 ±2.2 degrees, the control wheel force per "g" shall be increased to require an increasing force proportional to the pitch angle in excess of ±22 ±2.2 degrees.

4.2.6 Barometric Altitude Hold
The DAS shall be capable of operating in an Altitude Hold Mode.

The Altitude Hold Mode shall also be capable of entry from the Altitude Capture Mode when the selected altitude has been acquired.

During Altitude Hold Mode operation, the DAS Altitude Pitch Attitude Command shall be limited to ±22 ±2.2 degrees of pitch attitude and ±13 ±1.3° per second of pitch rate.

When the Autopilot Disconnect Switch is depressed, the autopilot shall disconnect, this state shall be indicated by a Caution Indicator.

In the altitude hold mode, the DAS shall be capable of stabilizing and maintaining the aircraft, in the pitch axis, to within FAA Reduced Vertical Separation Minima (RVSM) standards up to and including Flight Level 350.

The DAS shall provide caution and warning signals when the Altitude Hold Mode is engaged and the uncorrected barometric altitude deviates more than ±60 ±6 feet from the altitude at which the mode was engaged.

The warning indication shall include a repetitive sequence of aural tones.

Altitude Hold shall be augmented by a roll attitude signal to provide up elevator compensation in a turn.

4.2.7 Radar Altitude Hold

This mode shall operate systematically as the Barometric Altitude Hold function while within the range of the Radar altimeter (surface - 2,500 feet). No Radar Altitude Capture mode is required. This mode shall be visibly distinguishable from Barometric Altitude hold (an LED indicator is sufficient). Radar altitude hold shall be selectable at a minimum altitude of 200 feet above ground level and a maximum limit of the maximum radar altimeter range.

During Radar Altitude Hold Mode operation, the DAS IAS Pitch Attitude Command shall be limited to ±22 ±2.2 degrees of pitch attitude and ±13 ±1.3 degrees per second of pitch rate.

Terrain changes demanding control input outside of control law ranges or radar altimeter ranges shall illuminate a Caution indicator and allow system default to Attitude Hold mode.

Aircraft pitch down control input gains shall not produce descent rates, in feet per minute, greater than the radar altitude remaining, in feet Above Ground Level. In other words, a rate of descent shall not be generated so as to allow less than one minute to surface impact.

The DAS shall provide caution and warning signals when the Altitude Hold Mode is engaged and the uncorrected barometric altitude deviates more than ±60 ±6 feet from the altitude at which the mode was engaged.

The DAS shall be capable of stabilizing and maintaining the aircraft, in the pitch axis, to within ±50 feet of target altitude.

The warning indication shall include a repetitive sequence of aural tones.

Radar Altitude Hold shall be augmented by a roll attitude signal to provide up elevator compensation in a turn.

4.2.8 Indicated Air Speed (IAS) Hold

The DAS shall be capable of operating in an IAS Hold Mode, by controlling the pitch attitude of the aircraft, which shall hold the aircraft at the Indicated Airspeed existing at the time the mode is engaged ±10 ±2 KIAS.

During IAS Hold Mode operation, the DAS IAS Pitch Attitude Command shall be limited to ±22 ±2.2 degrees of pitch attitude and ±13 ±1.3 degrees per second of pitch rate. IAS Hold shall be augmented by a roll attitude signal to provide up elevator compensation in a turn.

4.2.8.1 Vertical Speed (VS) Hold

The DAS shall be capable of operating in a VS Hold Mode, by controlling the pitch attitude of the aircraft, which shall hold the aircraft at the indicated vertical speed existing at the time the mode is engaged ±100 ±50.

During VS Hold Mode operation, the DAS IAS Pitch Attitude Command shall be limited to ±22 ±2.2 degrees of pitch attitude and ±13 ±1.3 degrees per second of pitch rate. IAS Hold shall be augmented by a roll attitude signal to provide up elevator compensation in a turn.

4.2.9 Heading Select
The DAS shall be capable of operating in a Heading Select Mode.

The maneuver shall be smoothly executed, utilizing a coordinated turn which shall be bank angle limited to ±25 ±2.5 degrees and shall be command rate limited to ±5 ±0.5 degrees per second. When the aircraft approaches the pre-selected heading, the DAS shall control the aircraft to a wings level attitude in a non-abrupt manner.

4.2.10 Selected Altitude Arm and Capture
The DAS shall be capable of operating in an Altitude Capture Mode that shall command the aircraft to capture the altitude that has been selected by the pilot.

The Altitude Capture Mode shall provide a smooth transition from the climb/dive configuration to level flight, provided elevator authority is not exceeded due to engine power modifications. When the selected altitude has been acquired within ±9 feet, the Capture mode shall disengage and the Barometric Altitude Hold Mode shall become engaged.

While the Altitude Capture Mode is engaged, the pitch attitude shall be limited to ±22 ±2.2 degrees and the pitch rate shall be limited to ±13 ±1.3 degrees per second.

In Altitude Capture mode, normal acceleration shall be limited to ±0.2g from steady state acquiring the selected altitude from either a climb or descent.

4.2.11 Navigation Mode and Approach Mode
These Modes shall provide the ability to follow a vector in space based on vertical and lateral steering information provided over the ARINC 429 bus. The commanded pitch attitude authority shall be limited to ±5 ±0.5 degrees about the trimmed attitude and the commanded roll attitude authority shall be limited to ±35 ±3.5 degrees. The commanded pitch-rate authority shall be limited to ±13 ±1.3 degrees per second and the commanded roll-rate authority shall be limited to ±6 ±0.6 degrees per second.

In Approach mode, the DAS shall be capable of FAA certification to Instrument Landing System Category I criteria.

Missed approach disengagement is permissible however; some pilot cueing at missed approach is preferred.

4.2.12 Alternative NAV Mode
The DAS shall provide a dedicated NAV mode for coupling alternate external navigation signal systems (not specified). The availability of this mode shall be indicated by the presence of configuration jumpers input to the autopilot computer. The commanded pitch attitude authority shall be limited to ±5 ±0.5 degrees about the trimmed attitude and the commanded roll attitude authority shall be limited to ±35 ±3.5 degrees. The commanded pitch-rate authority shall be limited to ±13 ±1.3 degrees per second and the commanded roll-rate authority shall be limited to ±6 ±0.6 degrees per second.

4.2.13 Bank angle up elevator compensation
The DAS pitch modes shall be augmented by a term that is a function of roll attitude to provide up-elevator compensation in turns. The function of roll attitude that is being used is the "versine" of roll attitude (Φ) or $(1-\cos\Phi)/\cos\Phi$. This function is used to provide an up elevator command to compensate for the reduction of the vertical component of the lift vector when the aircraft is banked.

4.2.14 Automatic turn coordination
The DAS shall coordinate all possible turn maneuvers with lateral acceleration less than 0.05g maximum and less than 0.01g in steady state measured at the aircraft center of gravity. Turns commanded by the DAS shall be coordinated without the use of manual control in pitch and yaw.

4.2.15 Yaw damping
The DAS shall contain a redundant Yaw Damper. The Yaw Damper shall be engaged along with the autopilot.

The authority of the Yaw Damping System will be adequate to afford crew comfort and to provide good turn coordination.

The DAS shall provide a damping factor, zeta, of the Dutch roll mode of at least 0.7 for airspeeds in excess of 200 knots. For airspeeds from 1.3 Vs to 200 knots, the Yaw Damper shall provide a damping factor of at least 0.4.

4.2.15 Automatic pitch trim
The DAS shall provide electric motor servo commands to be used to reduce to a minimum the out-of-trim conditions produced by steady state changes in longitudinal trim introduced by changes in flight conditions. The DAS input signals shall be from the hydraulic load sensors on the elevator actuating cylinder. The pitch trim of the aircraft is controlled through the elevator trim-tabs.

Automatic pitch trim shall be operative when the autopilot is engaged. The trim tab surface shall be driven by an electrical servo actuator controlled by a servo amplifier in proportion to the hydraulic load at the elevator actuating cylinder.

4.2.16 Automatic pitch trim monitoring
A monitor shall be provided to compare the Trim Drive voltage at the motor against the calculated command that is expected from the load sensor output voltage. This monitor shall disengage the trim servo automatically when the miscompare occurs. Another monitor shall compare the outputs of the dual hydraulic load sensor and shall disengage the trim servo when the outputs do not agree. The disengagement of the trim servo by the monitors shall be indicated to the pilot.

4.2.17 Autopilot attitude limits
The DAS operation shall adhere to the following attitude limits unless stated otherwise:
a. Bank Angle: ±45 ±4.5 degrees
b. Pitch Angle: ±22 ±2.2 degrees

4.2.18 Switching transients
The DAS engagement and mode switching transients shall be no larger than ±0.1 g vertically and ±3 degrees per second laterally. The DAS shall synchronize the actuator commands of the three axes while disengaged so that upon engagement, there shall be no significant or undesirable motion of the manual controls.

4.2.19 Fail safe description
The DAS shall be a failsafe system. A fail-safe system is one that results in minimal aircraft transients in the event of any DAS malfunction.

The DAS shall utilize sensor inputs, where available, to provide the ability to detect failure or abnormalities in the DAS. Failures in the control surface actuators or their feedback signals shall cause no aircraft transients in excess of the following limits:
a. Pitch Axis - 0.25 g

4.2.20 Monitoring
The DAS shall include monitoring for the detection and isolation of system malfunctions with automatic system disengagement for detected failures. Aircraft transients resulting from the automatic disconnect shall be in accordance with the transient limits stated herein.

4.2.21 DAS authority limiting and overpower
The DAS control authority will be limited in each axis at the hydraulic servo. The servo limits shall be supplemented by electronic limits of the servo commands. Manual pilot overpower shall be possible through the authority mechanism at reasonable force levels.

4.2.22 Safety

The DAS shall warn the flight crew of any flight safety critical failures with sufficient margin so that a safe recovery of the aircraft is possible.

The DAS shall meet the following safety requirements:
a) With the Autopilot engaged, the aircraft shall not exhibit any uncommanded deviations from flight path without crew warning and automated disconnect.
b) The Autopilot shall be able to hold the aircraft throughout the flight envelope as defined in the P-3 Orion Flight Manual, without any uncommanded maneuvers.
c) The Autopilot shall not exhibit any control input resulting in acceleration such that an aircraft overstress condition is produced.
d) The Autopilot shall not be allowed to position the aircraft in an attitude from which a recovery is either impossible or requires aircraft overstress.

4.2.22.1 Visual Warnings
Warnings shall be provided to indicate non-manual autopilot disengagement resulting from any conditions.

4.2.22.2 Aural Warnings
The aural tone generator shall be energized for any condition that warrants immediate pilot attention. Aural warnings may be used for pilot cueing of autopilot mode changes or functionality degraders.

4.2.23 Attitude sources
The normal state of the DAS shall be to use the average of INS 1 and INS 2 attitude data in each channel. It shall also be possible for the DAS to use a single INS source.

4.2.24 Barometric Altitude Correction
The DAS shall provide the capability for barometric altitude correction.
Corrected barometric altitude shall be used in the altitude capture mode. The pressure shall be adjustable from 22 in. Hg to 32 in. Hg (or equivalent) and stored in non volatile memory. A default value of 29.92 in. Hg shall be used until updated by the crew. The resolution of the value shall be 0.01 in. Hg

4.2.25 Watchdog timer
A timer shall be incorporated into the digital processor design to monitor the processor for the proper cyclic operation. The timer output shall be used to provide failure indications and prevent autopilot engagement when not stimulated at a regular rate by the processor.

4.2.26 Initialization
DAS initialization shall occur automatically upon application of electric power or as the result of recovery from loss of power. After power application, the engagement of the DAS shall be inhibited for a reasonable period of time (not to exceed 10 seconds) to ensure the proper state of the system is achieved.

4.2.27 Failure disconnect logic
The DAS shall contain hardware logic within the watchdog timer to prevent a failed processor from being engaged and commanding the control surfaces.

4.3 Growth allowances
The DAS must reserve capability for future system growth.

4.3 Processing growth allowances
The average throughput of time critical tasks shall not exceed 80% of the available processing time. This will allow for future growth.

4.3.1 Memory growth requirements
The DAS shall utilize a maximum of 60% of the available memory on the baseline system. This will allow for future growth.

4.4 Built-In Test (BIT)
The BIT function shall have the following design goals:
a. Detect 97% of failures.
b. Isolate 95% of failures detected to a single Shop Replaceable Assembly (SRA).
c. False alarm rate of less than 6%.

4.5 Operational safety
Operational safety features shall include both hardware and software monitors that disengage the Autopilot when a particular failure renders the Autopilot incapable of proper operation or renders proper system operation in question.

4.6 Software requirements
4.6.1 Introduction
The Contractor shall demonstrate that the software has been designed, developed, tested and certified in a manner that ensures software performance, integrity and supportability throughout its life and the life of the associated hardware systems.

4.6.2 Critical software
The Contractor shall certify that the DAS software has been designed, tested, and integrated IAW recognized international standards for safety critical software. The software verification effort shall be guided by the process requirements of RTCA/DO-178B Level C. Testing shall be focused at a system level with traceability to the constituent software components

4.6.3 Quality system
The Contractor shall develop and deliver software and associated software deliverables identified in the SOW IAW the software quality system ISO 9000.3.

4.6.4 Software reliability, maintainability and usability

The Contractor shall design new and modified re-used software to ensure through life reliability, maintainability, and usability. No software fault shall cause the failure of a mission or a safety critical capability.

4.6.5 Viruses
The Contractor shall ensure that all software and firmware, which has been used in the development of the DAS software, is free of viruses, locks and drop-dead devices prior to delivery to NASA.

4.7 Electromagnetic Interference (EMI) and Electromagnetic Compatibility (EMC) requirements
In order to reduce costs, the contractor shall use existing test data from similar DAS installations to demonstrate compliance with the following EMI/EMC requirements to the maximum extent.

4.7.1 Electromagnetic environmental effects
The modification of the aircraft and installation of the DAS, including new and modified retained components, shall conform to the requirements of MIL-STD-464 (Electromagnetic Environmental Effects – Requirements for Systems).

The Contractor shall demonstrate that the electromagnetic environmental effects associated with the modification of the aircraft and operation of the DAS does not interfere with or degrade the performance and function of existing aircraft systems.

The Contractor shall demonstrate that the electromagnetic environmental effects associated with the operation of existing aircraft systems does not interfere with or degrade the performance and function of the DAS.

The Contractor shall demonstrate that the electromagnetic environmental effects associated with external emitters (e.g., onboard emitters) does not interfere with or degrade the performance and function of the DAS.

4.8 Environmental requirements
The equipment shall deliver specified performance when subjected to the environmental conditions specified within and shall be tested IAW the sections of MIL-STD-810F, as detailed in this section. In order to reduce costs, the contractor shall use existing test data from similar DAS installations to demonstrate compliance with the following requirements, to the maximum extent.

4.8.1 Combined temperature altitude
The equipment shall deliver specified performance when subjected to combined temperature-altitude environment as specified below.

4.8.1.1 Temperature

The equipment shall deliver specified performance when subjected to temperature environment IAW MIL-STD-810F, Method 520. 1, Procedure 111, over the ambient temperature range, DAS Equipment Temperature Environment, for all components of the DAS system:

	Operating	Non-Operating
All DAS Components and Sub-Systems (degrees C):	-25 to +50	-30 to +70

4.8.1.2 Altitude

The equipment shall deliver specified performance when subjected to altitude environment IAW MIL-STD-810F, Method 520.1, Procedure 111, at a simulated altitude of 36,000 feet.

4.8.2 Vibration

The equipment shall deliver specified performance when subjected to vibration environment IAW MIL-STD-810F, Method 514.4, Category 4, with a sweeping sinusoidal excitation over the frequency range of 5 to 500 Hz, with double amplitude of 0.1 inches from 5 to 20 Hz and from +2g to –2g between 20 and 500 Hz.

Resonant modes at 60 to 76, 120 to 152, 188 to 220, and 255 to 290 Hz shall be minimized.

4.8.3 Shock

The equipment shall be constructed to withstand shock testing IAW MIL-STD-810F, Method 516.4, procedures I (Functional) and V (Crash Safety) IAW the following paragraphs. Each shock condition shall have a time duration of 11±1 ms.

4.8.3.1 Shock (Functional)

The equipment shall not suffer damage or subsequently fail to provide the performance specified when subjected to 18 impact shocks of 10g, consisting of three shocks in opposite directions along each of three mutually perpendicular axes.

4.8.3.2 Shock (Crash)

The equipment shall withstand 12 impact shocks of 20g, consisting of two shocks in opposite directions along each of three mutually perpendicular axes with no failure in attaching joints. Bending and distortion of the equipment shall be permitted.

4.8.4 Sand and dust

The equipment shall be constructed to withstand Sand and Dust testing IAW MIL-STD-810F, Method 510.3. The Control Panel is exempt from this requirement.

4.8.5 Salt fog

The equipment shall be constructed to withstand Salt Fog testing IAW MIL-STD-810F, Method 509.3 with exposure to an atomized 54-11% salt fog solution for 48 hours.

4.8.6 Humidity and moisture

The equipment shall be constructed to withstand Humidity and Moisture testing IAW MIL-STD-810F, Method 507.3 at 100% relative humidity.

4.8.7 Fungus
The equipment shall be constructed to resist Fungus growth. Fungus inert material IAW MIL-STD-810F, Method 508.4 shall be used.

4.9 DAS physical characteristics
4.9.1 Handles
Large components (greater than 20 lbs) should incorporate a handle(s) for installation, removal, and carrying.

4.10 Thermal design
The equipment thermal design shall follow the guidelines of MIL–HDBK–5400 for class II equipment except as modified by this specification.

4.11 Cooling
The DAS components shall be designed to meet performance requirements of this specification without cooling air. Under normal operating conditions, the system should operate minimally 10% below manufacturer's maximum operating temperature.

4.12 Electrical power
The Contractor shall install a DAS that shall maintain or reduce the overall autopilot power requirements.

4.12.1 115 Vac primary power
The installation of the DAS shall not reduce the quality of the aircraft electrical power system. The DAS components shall meet the specified performance when supplied 115 Vrms, 400 Hz ac power in accordance with MIL-STD-704A. The equipment shall meet all steady state requirements of MIL-STD-704A for the types of power specified herein, and shall give specified performance when supplied with electrical power having characteristics and limits as defined in MIL-STD-704, Category A, except for the requirement for normal operation during and after 50 ms of power interruption, where the DAS shall meet as a minimum 2 ms of power interruption. Whenever the supply voltage is too low to provide reliable processor operation, the processor driven outputs of the DAS shall be deactivated. There shall be no erratic servo commands due to abnormally low primary input power. Should the power source not meet operating requirements, the DAS shall be shut down. The installation of the DAS shall not further reduce the quality of the aircraft electrical power system. The Contractor shall carry out an Electrical Load Analysis in accordance with MIL-E-7016F for all new and modified systems and provide a copy of the analysis to NASA as outlined in the SOW. The DAS may also be supplied with 28 Vdc per MIL-STD-704A to be used as needed to interface with aircraft equipment.

4.12.2 Power consumption

Total DAS Power Consumption shall not exceed that of the existing PB-20N Autopilot system (estimated 350 watts).

4.12.3 Power interruptions
The Contractor shall design the DAS to be tolerant of the aircraft and ground power electrical systems. This requirement includes power supply transients and the ability to operate normally without damage, after voltage and frequency surges and transients in the AC and DC electrical power systems (e.g., Generator switching).

The equipment shall not be damaged by power interruption for any time period. During a power interruption, the computer shall undergo an orderly shutdown. This shutdown requires that the servo commands shall be disengaged. When power returns, the equipment shall automatically restart and be ready to be engaged by the pilot within a time period of 10 seconds.

4.13 Design and construction
The design and construction of the equipment shall use MIL-HDBK-5400 as a guideline. Maximum consideration shall be given to standardization, reliability, maintainability, EMI/EMC, serviceability, safety, and logistics to obtain the service performance of the equipment.

4.13.1 Materials, processes and parts
Multi-source parts shall be used wherever possible. The materials, processes, and parts shall use MIL-HDBK-5400 paragraph 3.1 as a guideline, wherever feasible.

4.13.2 Relays
The use of electromechanical relays shall be limited to those applications that require characteristics of high electrical isolation, defined unpowered state, or current handling capability. All relays shall have internally or externally arc suppressed coils.

4.13.3 Electrostatic protection of parts
Electronic parts that can be damaged by 2000 volts or less Electrostatic Discharge (ESD) shall be protected during manufacturing and delivery. Printed wiring assemblies containing ESD devices will be marked with ESD handling caution. All drawings shall identify ESD sensitive components.

4.13.4 Coatings
Printed circuit boards and other components shall be conformal coated as required to withstand environmental conditions. Coatings shall be capable of removal during repair without damage to the components.

4.13.5 Internal electrical connectors
Internal electrical connectors shall be polarized so that improper connections are impossible.

4.13.6 Shielding and bonding on finished surfaces
Non-conductive oxides or other non-conductive finishes shall be removed from the actual contact area of all surfaces required to act as a path for electric current and from local areas to provide continuity of electrical shielding or bonding. All mating surfaces shall be clean and shall be carefully fitted, as necessary, to minimize radio frequency impedance at joints, seams, and mating surfaces.

4.13.7 Mechanical installation and restraint
Provisions for installation and restraint in the aircraft shall comply with SOW.

4.13.8 Nameplates and Product Marking
4.13.8.1 Identification of Equipment
All equipment shall be identified with the manufacturer's name, part number, serial numbers and modification revision.

4.13.8.2 Nameplates
Nameplates and product marking shall be similar to the requirements of MIL-STD-130L as appropriate for the type and size of the product.

When the item is too small to accommodate a nameplate, the product shall be identified by permanent type marking applied directly to the part using any suitable means authorized by MIL-STD-130L.

4.13.8.3 Nomenclature
Nomenclature shall follow the guidelines of MIL-HDBK-5400 paragraph 3.4.

4.13.9 Workmanship
Workmanship shall follow the guidelines of MIL-HDBK-5400 paragraph 3.5.

4.13.10 Interchangeability
Interchangeability shall exist between all units and replaceable assemblies, subassemblies, and parts for all equipment delivered on the contract shall follow the guidelines of MIL-E-5400 paragraph 3.3.

4.13.11 Safety
The DAS shall incorporate design features that will not present health and safety hazards to personnel who will handle the equipment under MIL-STD-882D.

4.13.12 Human performance/human engineering
Human engineering shall be IAW MIL-STD-1472F.

4.14 Aircraft requirements
The DAS shall be integrated with the existing systems on the P-3 Orion aircraft and be capable of future expansion. With exception to the PB-20N Autopilot system, current

display systems, and flight management computers, NASA wishes to minimize changes to the existing systems fitted to the P-3 Orion aircraft.

4.14.1 Weight and balance

The installation of the DAS shall not increase the basic weight of the baseline P-3 Orion aircraft. The Contractor shall aim to reduce weight wherever possible. The Contractor shall perform a weight and balance analysis for the modified aircraft and provide a copy of the analysis NASA as a part of the Final Data Package.

4.14.2 Aircraft center of gravity

The installation of the DAS shall not move the aircraft center of gravity such that the new baseline P-3 Orion aircraft center of gravity exceeds the limitations outlined in the P-3 Orion Flight Manual.

4.14.3 Hydraulic power

The DAS shall interface with the existing aircraft hydraulic systems in such a way that engineering changes are minimized. The Number 1 hydraulic system on the P-3 Orion aircraft supplies pressure to control Autopilot engagement and to control the flight control booster packages.

The existing Autopilot may be engaged when the hydraulic system pressure exceeds 1800 psi and, via an engage circuit interlock, shall disengage if the system pressure to the booster packages drops below 1100 psi.

4.14.4 Colors and markings

4.14.4.1 Paint scheme

The Contractor shall finish all new, modified and newly exposed surfaces in a color that match the surrounding surfaces paint scheme (gray or black). Any markings removed during modification are to be re-applied.

4.14.4.2 Aircrew station labels and legends

The Contractor shall apply aircrew station labels and legends that comply with the requirements of MIL-STD-1472F.

4.15 Engineering requirements

As a standard for NASA aircraft modification - NAVAIR 01-1A-505 shall be used. This standard covers the design and modification of aircraft structural, mechanical, and electrical components and systems. Where conflicts exist between NAVAIR 01-1A-505 and published NASA aircraft modification standards, the more restrictive standard shall be used.

The Contractor shall demonstrate that all engineering changes and aircraft modifications meet the requirements of NAVAIR 01-1A-505, and published NASA standards, unless the Contractor can demonstrate to NASA's satisfaction that this is inappropriate or uneconomical.

The Contractor shall design and develop the DAS IAW a recognized international standard that ensures equipment performs its intended function; equipment is protected against failure conditions; and the operation or failure of the equipment does not adversely effect the operation of other systems. The NASA's preferred standard for the design of highly integrated aircraft systems is MIL-F-9490. The DAS shall use the design, performance, and testing requirements of MIL-F-9490 as a guide for performance evaluation during the flight-test phase.

The DAS shall not have any adverse effect or induce maneuvers that may cause any adverse effect on the structural integrity and fatigue life of N426NA (P-3 Orion aircraft). The Contractor shall provide full documentation of all modifications and changes that are made to the aircraft structure and systems during the installation of new equipment and removal or relocation of existing equipment. The detail shall be sufficient to demonstrate the integrity of the modified installation.

4.15.1 Materials and processes
The Contractor shall select materials and processes that meet the requirements stated below and maximize the protection and resistance to corrosion. The NASA's specific requirements are detailed below.

4.15.1.1 Materials
Only materials listed in MIL-HDBK-5J shall be used in airframe and aircraft structural modification.

4.15.1.2 Protective coatings
The Contractor shall apply protective coatings to provide new and modified retained equipment, systems, and structure with protection from corrosion, abrasion, and other deleterious action.

4.15.1.3 Aluminum components
The Contractor shall treat and apply corrosion protective coatings to all aluminum components as follows:
a. All new and modified aluminum panels and structural components shall be treated with a chemical conversion (Alodine 1200).
b. All new and modified interior aluminum panels and structural components that are not subject to electrical bonding requirements shall have two coats of primer.
c. All new and modified exterior aluminum panels and structural components shall be primed and painted with one coat of primer and two coats of polyurethane enamel.
d. Sliding aluminum surfaces shall be treated IAW either Type I or Type II Anodic coating.

4.15.1.4 Steel components
The Contractor shall treat and apply corrosion protective coatings to all steel components as follows:

a. Alloy steel components with a tensile strength greater than 180 ksi shall be vacuum cadmium plated; aluminum coated; or otherwise treated.
b. Alloy and carbon steel components with tensile strength less than 180 ksi shall be cadmium plated.
c. All alloy steel parts shall be primed and painted with one coat of primer and two coats of polyurethane enamel.

4.15.1.5 Fiberglass laminate and composite components
The Contractor shall comply with the following requirements when using fiberglass laminate and composites:
a. All new and modified interior fiberglass surfaces shall have a resin coat applied to seal the surface.
b. All new and modified exposed interior fiberglass surfaces shall have two coats of MIL-PRF-85285E color.
c. All new and modified exposed exterior laminates shall be coated to preclude rain erosion damage and to provide anti-static protection.

Sealants
The Contractor shall comply with the following sealant requirements when modifying the aircraft:
a. All exterior structural members, joints and seams and those in interior areas exposed to corrosive conditions shall be sealed.
b. All fasteners shall be installed wet using sealant. The P-3 Orion Structural Repair Manual (NAVAIR 01-75PAA-3-1) defines the process for the wet installation of fasteners.
c. All cured sealant in exterior locations shall be protected by the appropriate exterior finish scheme.

4.15.2 Electrical wiring and equipment
4.15.2.1 Wiring standard
The Contractor shall comply with the requirements of MIL-W-5088L for aircraft wiring, wire marking, and wiring processes.

4.15.2.2 KAPTON™ wiring
The NASA has aggressively attempted to remove MIL-DTL-81381B standard wiring or any wiring employing KAPTON™ or Polyamide type insulation from its P-3 Orion aircraft. The Contractor shall not use MILDTL- 81381B standard wire or any other wire employing KAPTON™ or Polyamide insulation during the modification of the P-3 Orion aircraft or installation of new equipment. Where the Contractor is required to modify a wiring loom that contains MIL-DTL-81381B standard wire or any other wire employing KAPTON™ or Polyamide insulation, the Contractor shall replace it with MIL-W-5088L wire.

4.15.2.3 Polyvinyl Chloride (PVC) insulated wiring
The use of Polyvinyl Chloride (PVC) insulated wiring is prohibited.

4.15.2.4 Redundant wiring looms
NASA requires redundant wiring to be removed during aircraft modification to reduce aircraft weight and the possibility that it will act as RF re-radiators. The Contractor shall remove redundant wiring and/or wiring looms associated with removed or relocated equipment.

4.15.2.5 Circuit breakers
Circuit breakers shall be of the fault indicating type and be functionally grouped and located on existing Circuit Breaker Panels. All new and relocated Circuit Breakers shall be uniquely numbered and clearly marked with rating, size and function.

4.15.2.6 Fuses
The use of fuses shall be minimized and confined to the unfiltered 28 Vdc supply.

4.15.2.7 Overload protection
The Contractor shall provide short circuit and overload protection.

4.15.2.8 Sensitive Electronic Device (SED)
The Contractor shall mark all SED items with the SED symbol.

4.15.2.9 Terminal boards
The Contractor shall uniquely number all new and relocated terminal boards..

4.15.3 Safety
The Contractor shall implement a System Safety Program that is appropriately tailored from and meets the intent of MIL-STD-882D.

4.15.3.1 Occupational health and safety
The Contractor shall design the new and modified retained equipment so that it does not introduce a hazard to the health or safety of personnel handling, operating or maintaining it. This is a life cycle requirement that exists whether the equipment is operating, installed but not operating, being fitted or removed, under repair or maintenance, or in storage.

4.15.3.2 Hazardous substances
The Contractor shall avoid the use of hazardous substances, including Halons and Carbon Flourochlorides (CFCs), unless specifically authorized by NASA.

The Contractor shall not introduce new or modified retained equipment or components that use substances banned under US law or substances that can cause occupational injury or disease.

4.15.3.4 Use of fiberglass laminates and composites

Where the Contractor introduces fiberglass laminate and composite materials into the aircraft interior the Contractor shall provide measures to minimize the introduction of fiberglass and composite fibers into the cabin environmental system.

4.15.3.5 Voltage warning plates

The Contractor shall mark all new and modified retained equipment that contains potential between 70 and 500 volts with warning signs or labels IAW ANSIZ35.1 Class II and ANSIZ35.4, and shall read as a minimum "Caution – (insert maximum voltage applicable) Volts." For potentials in excess of 500 volts the Contractor shall use warning signs IAW ANSIZ35.1 Class I and ANSIZ35.4, and shall read as a minimum "DANGER – HIGH VOLTAGE – (insert maximum voltage applicable) VOLTS".

4.16 Reliability and maintainability

Equipment maintenance provisions shall conform to requirements of this specification and the equipment shall meet the following quantitative and qualitative requirements.

4.16.1 Reliability

The Mean Time Between Failure (MTBF) of the DAS shall be at least 3,000 flying hours. The Contractor shall provide component and system reliability and MTBF data for the DAS and state whether the quoted MTBF is historic (field) or predicted, and if predicted using MIL HDBK-217F, whether any de-rating has been applied.

The Contractor shall warrant the reliability performance stated for the first 12 months of service or as outlined in the SOW, whichever is more advantageous to the government.

The Contractor shall demonstrate that the introduction of new and modified retained equipment does not degrade the reliability of retained aircraft equipment (based on pre-modification base-line reliability performance).

4.16.2 Maintainability

The design of the equipment shall be such that its maintenance shall not require skills that exceed the capability of a trained maintenance technician.

4.16.2.1 Organization-level testability

The DAS shall be designed such that no flight line Peculiar Support Equipment (PSE) shall be required for checkout upon installation, for confidence checks, or for fault isolation to the DAS.

4.16.3 Organizational level maintainability requirements

4.16.3.1 Equipment handling

The equipment shall be designed and constructed such that on-aircraft maintenance can be performed in environments of any humidity (up to 100% relative humidity), temperatures of –40°C to +71°C, and specified sand and dust, by personnel wearing clothing required by the particular environment such as heavy gloves. Required maintenance such as removal, replacement, and hook-up shall be possible over this expected range of flight-line environments with only external cleaning or wiping allowed.

4.16.3.2 Adjustment, alignments, and calibrations
"On-aircraft" maintenance adjustment, alignments, or calibrations shall not be required for this equipment.

4.16.3.3 Aircraft ready condition
Equipment, when installed in an aircraft that has been released for flight, shall be capable of meeting the performance requirements specified herein after a period of at least 24 hours during which no maintenance, checkout, or flight has occurred.

4.16.4 Reversibility restrictions
The equipment design and construction shall incorporate features such that it is mechanically and electrically impossible to install components incorrectly or to attach electrical plugs or mechanical restraints in an improper manner.

4.16.5 Preventative maintenance
There shall be no scheduled maintenance (including maintenance inspections and parts replacement) required for the equipment.

4.16.6 Accessibility
The equipment shall be designed and constructed such that it shall be possible to remove and replace any new DAS component, other than a motherboard, without removing or disconnecting any other assembly.

5.1 QUALITY ASSURANCE REQUIREMENTS
The Contractor is responsible for the performance of all inspection, analysis, and test requirements as specified herein and the implementation of an effective and economical quality control program.

5.2 Formal qualification
The following paragraphs specify the requirements for, and methods of, formally verifying that the DAS requirements have been satisfied. Functional testing may be required both during and/or after qualification tests.

Requirements may be verified by one method or a combination of methods. Verifications are categorized into the following three types according to the method of accomplishment:
a. Inspection
b. Analysis/Similarity
c. Test

5.2.1 Inspection

The requirements shall be verified by inspection of the assembly, applicable drawings, and assembly records at the time and place of qualification testing.

5.2.2 Analysis/Similarity
The following requirements shall be verified by review and approval of analytical data or the device's similarity to another qualified unit.

5.2.2.1 Reliability
Compliance with the numerical requirements shall be demonstrated by quantitative analysis, IAW MIL-HDBK-217F.

5.2.2.2 Maximum "g" loads
Maximum load factors shall be based on one of two methods:
a. Design
b. Analysis

5.2.3 Test
Physical and electrical tests shall be conducted IAW the requirements in this specification.

5.2.3.1 Test criteria
Satisfactory performance is defined as operations in which the system performs electrically and mechanically within the specification requirements when operated IAW this specification.

5.2.3.2 Test equipment
It shall be the responsibility of the Contractor to establish the accuracy requirements for measurement of the various parameters during any test unless otherwise specified herein.

5.2.3.3 Test conditions
Unless otherwise specified, all tests shall be performed at standard laboratory conditions defined as follows:
a. Atmospheric pressure of 24.0 to 32.0 in. Hg;
b. Temperature of 25 ± 5°C; and
c. Relative humidity of 80% or less.

Whenever voltages are not specified for test conditions, they shall be assumed to be the nominal value.

5.2.3.4 Test results
All recording of test results shall be quantitative in nature i.e. actual test results shall be recorded. If a PASS/FAIL test is utilized, a PASS/FAIL criteria must be specified.

5.3 System Acceptance Test Plan

The Contractor will submit the System Acceptance Test Plan (SATP) for approval prior to the article acceptance tests in accordance with the timetable established in the SOW.

SATP flights will be conducted by a NASA specified crew. Flight Crew work load will be assessed by means of the Bedford Workload scale.

5.3.1 Human-Machine Interface

NASA requires flight crew interactions with interfaces and all tasks required of the flight crew shall be designed to meet a workload rating of 3 or better on a Bedford Workload Scale, or the Modified Cooper-Harper Scale, or equivalent workload scales used to evaluate flight crew workload in accordance with NPR 8705.2

APPENDIX B
NNG09299178R

Data Outputs Available to P-3B User

PADS High Speed Arinc 429 Channel

This channel will provide Navigation Data originating from the FMS #1, FMS #2, INS #1, INS #2, ADC #1, ADC #2, GPS, or any additional navigation system or aircraft sensor installed. The data to be transmitted will be user selectable. The word labels for PADS output are tentatively defined below. Each 32 bit data word will be followed by its corresponding 32 bit time tag. The time tag transmission may be turned off if requested.

PADS RS-232 Channel

This channel will provide Navigation Data originating from the FMS #1, FMS #2, INS #1, INS #2, ADC #1, ADC #2, GPS, or any additional navigation system or aircraft sensor installed. The data to be transmitted and the baud rate will be user selectable. The word format is to be determined. A data word may be followed by its corresponding time tag if requested.

PADS RS-232 Time Update Channel

A single line devoted to time transmission will be available in order for the user(s) to synchronized their instruments. The time transmitted will originate from the GPS, and will have an accuracy of 1 ms.

Note: The following information was researched by Evan Webb of Code 831.1.

Honeywell LaserRef III Inertial Reference Systems (IRS1 and IRS2) HS Arinc 429 Channels

PADS Label	Label	Arinc Data	Resolution	Accuracy	Units	Positive Sense	Updates
001	310	Latitude	.00017172	*	deg	CW from N	5 Hz
002	311	Longitude	.00017172	10	deg	CW from N	5 Hz
003	312	Ground Speed	.0039	12	kts	Always Pos.	10 Hz
004	313	Track Angle True	.00017172	5.0	deg	CW from N	25 Hz
005	314	True Heading	.00017172	0.4	deg	CW from N	25 Hz
006	315	Wind Speed	.000224	12	kts	Always Pos.	10 Hz
007	316	Wind Direction True	.00017172	10	deg	CW from N	10 Hz
008	317	Track Angle Magnetic	.00017172	6	deg	CW from N	25 Hz
009	320	Magnetic Heading	.00017172	*	deg	CW from N	25 Hz
010	321	Drift Angle	.00017172	5	deg	Right	25 Hz
011	322	Flight Path Angle	.00017172	0.4	deg	CW from N	25 Hz
012	323	Flight Path Acceleration	.000122	10%	g	Forward	50 Hz
013	324	Pitch Angle	.00017172	0.1	deg	Up	50 Hz
014	325	Roll Angle	.00017172	0.1	deg	Rt. Wing Down	50 Hz
015	326	Body Pitch Rate	.0039	0.1 or 1%	deg/s	Up	50 Hz
016	327	Body Roll Rate	.0039	0.1 or 1%	deg/s	Rt. Wing Down	50 Hz
017	330	Body Yaw Rate	.0039	0.1 or 1%	deg/s	Nose Right	50 Hz
018	331	Body Long. Accel.	.000122	0.01	g	Forward	50 Hz
019	332	Body Lateral Accel.	.000122	0.01	g	Right	50 Hz
020	333	Body Normal Accel.	.000122	0.01	g	Up	50 Hz
021	334	Platform Heading	.00017172	0.4	deg	CW from N	25 Hz
022	335	Track Angle Rate	.0000305	.25	deg/s	CW	50 Hz

023	336	Pitch Attitude Rate	.0039	0.1 or 1%	deg/s	Up	50 Hz
024	337	Roll Attitude Rate	.0039	0.1 or 1%	deg/s	Rt. Wing Down	50 Hz
025	360	Potential Vertical Speed	1	30	ft/min	Up	50 Hz
026	362	On-Track Hor. Accel.	.000122	10%	g	Forward	50 Hz
027	363	Crosstrack Hor. Accel.	.000122	10%	g	Right	50 Hz
028	364	Vertical Acceleration	.000122	0.01	g	Up	50 Hz
029	365	Inertial Vertical Speed	.03125	30	ft/min	Up	25 Hz
030	366	North-South Velocity	.0039	12	kts	North	10 Hz
031	367	East-West Velocity	.0039	0.1 or 1%	deg/s	East	10 Hz

*Computed between latitudes 73N and 60S only. Accuracy between 50N and 50S: 2 deg. Accuracy at greater than 50 deg latitude: 3 deg. Maximum of 15 deg/hr drift.

Universal Navigation UNS-1B Flight Management System (FMS1 and FMS2) HS Arinc 429

PADS Label	Label	Arinc Data	Resolution	Accuracy	Units	Positive Sense	Updates
050	115	True Waypoint Bearing	.00005464	**	deg	CW	10 Hz
051	116	Cross Track Distance	.000122	**	NM	Right	10 Hz
052	147	Magnetic Variation	.00005464	**	deg	CW	2 Hz
053	150	Greenwich Mean Time	1 sec	**			1 Hz
054	163	Wind on Nose	.0002441	**	kts	Headwind	2 Hz
055	174	Pseudo Glideslope Dev.	.002	**	ft	Up	10 Hz
056	203	Pressure Altitude	.125	**	ft	MSL	2 Hz
057	204	Altitude	1	**	ft	MSL	5 Hz
058	210	True Airspeed	.0625	**	kts	Always Pos.	2 Hz
059	213	Static Air Temperature	.0004883	**	deg C		1 Hz
060	306	Waypoint Latitiude	.00017166	**	deg	North	10 Hz
061	307	Waypoint Longitude	.00017166	**	deg	East	10 Hz
062	310	Latitude	.00017166	**	deg	North	10 Hz
063	311	Longitude	.00017166	**	deg	CW from N	10 Hz
064	312	Ground Speed	.125	**	kts	Always Pos.	5 Hz
065	313	True Track Angle	.00005464	**	deg	CW	2 Hz
066	314	True Heading	.0055	**	deg	CW from N	10 Hz
067	315	Wind Speed	.0002441	**	kts	Always Pos.	2 Hz
068	316	Wind Angle	.00005464	**	deg	CW	2 Hz
069	321	Drift Angle	.00005464	**	deg	CW	10 Hz
070	351	Distance to Destination	.03125	**	NM	Always Pos.	2 Hz
071	352	Time to Destination	.015625	**	min	Always Pos.	2 Hz
072	366	North-South Velocity	.0039	**	kts	Always Pos.	5 Hz
073	367	East-West Velocity	.0039	**	kts	East	2 Hz

** Accuracy of parameters which are passed through the FMS are equal to the output accuracy of the equipment from which the data are input. Accuracy of the FMS itself is difficult to quantify since the parameters which are computed by the FMS are based on a Kalman filter solution of present position which has time-varying coefficients. However, the FMS has been shown to introduce no additional computational inaccuracy due to such sources as insufficient internal data word length or roundoff error, etc.

Sperry AZ-810 Digital Air Data Computer (ADC1 and ADC2) LS Arinc 429

PADS Label	Label	Arinc Data	Resolution	Accuracy	Units	Positive Sense	Updates
075	203	Pressure Altitude	1	200	ft	Always Pos.	1 Hz
076	204	Baro. Corrected Alt.	1	5(correction)	ft	Always Pos.	1 Hz
077	205	Mach Number	.0000625	.01	Mach	Always Pos.	1 Hz
078	206	Indicated Airspeed	.0625	4	kts	Always Pos.	1 Hz
079	207	Vmo (Max. Op. Speed)	.25	4	kts	Always Pos.	2 Hz
080	210	True Airspeed	.0625	4	kts	Always Pos.	4 Hz

081	211	Total Air Temperature	.25	1	deg C		1 Hz
082	212	Altitude Rate	16	30 or 5%	ft/min	Up	16 Hz
083	213	Static Air Temperature	.1	1	deg C		1 Hz

Universal Navigation GPS/OSS GPS Receiver LS Arinc 429

PADS Label	Label	Arinc Data	Resolution	Accuracy	Units	Positive Sense	Updates
100	076	Altitude	.125	***	ft	MSL	900ms
101	103	Track Angle	.0055	***	deg	CW	900ms
102	111	Latitude	.000172 ***	deg	North	900ms	
103	112	Longitude	.000172 ***	deg	East	900ms	
104	116	Ground Speed	.125	***	kts	Always Pos.	900ms
105	120	Cross Track Distance	.004	***	NM	Right	2 Hz
106	124	Measurement Age	.05	***	sec	Always Pos.	900ms
107	125	GMT (UTC)	min/10	***	hr,min	Always Pos.	2 Hz
108	136	Vertical FOM	.031	***	m	Always Pos.	900ms
109	140	UTC Fine	.95367us	***	sec	Always Pos.	900ms
110	151	UTC	1 sec	***	s,min,hr	Always Pos.	900ms
111	165	Vertical Velocity	1	***	ft/min	Up	900ms
112	226	Latitude FOM	.031	***	m	Always Pos.	900ms
113	227	Longitude FOM	.031	***	m	Always Pos.	900ms

*** GPS accuracy in non-differential applications for a receiver using the Coarse Acquisition (CA) signal is limited to 35m when Selective Availability (SA) is turned off and 100m when SA is on. This assumes that the satellite geometry is favorable and that four satellites are in view: otherwise, accuracy will be degraded.

For Additional Information or Requirements Contact: George L. Jackson (804) 824-2390 or Evan Webb (804) 824-1375.

APPENDIX C - NNG09299178R

NASA 426 P3B BUNO: 152735 AVIONICS EQUIPMENT

NOMENCLATURE	QTY	MODEL	MFR.	PART NUMBER
EFIS EQUIPMENT				
EFIS DISPLAYS	5EA	EDZ-805	SPERRY	7003110-902
SYMBOL GENERATOR	2EA	SG-805	SPERRY	70116272-807
MULIT-FUNCTION SYMBOL GENERATOR	1EA	MG805	SPERRY	70116273-807
MFD CONTROL	1EA	MC-800	SPERRY	7007062-922
DISPLAY CONTROLLER	2EA	DC-811	SPERRY	7012977-708
EFIS FAN	3EA	718ZH	ROTRON	26977
SENSOR-AIR FLOW	3EA		OMEGA	FSW-112R
REMOTE COURSE HDG.	2EA	RI-1065	SPERRY	4026206-961
INCLINOMETER	2EA	AY-003	SPERRY	7005400-902
EFIS INTERFACE	2EA	12 POLE RELAY	K&S	1001-003
F/D COMPUTERS	2EA	FZ500	HONEYWELL	4018369-905
MODE SELECTOR	1EA	MS-500A	HONEYWELL	7004536-914
MOMENTARY SWITCH CONTROL	1EA		CONT.SWITCH COM.	C2006B
MOMENTARY SWITCH	1EA		CONT.SWITCH COM.	B9021BR
ACCELEROMETER	1EA		HONEYWELL	7000992
FMS-NCU	2EA	UNS-1B	UNC	1190-01-1111
FMS-CDU	1EA		UNC	1016-1-21
FMS RADIO TUNE UNIT	1EA		UNC	1056-01-11
FMS SWITCH PACK UNIT	1EA		K&S	1003-003
DTU-DATA TRANSFER UNIT	1EA		UNC	1405-01-2

P-3B Orion (N426NA)

P-3B Design Requirements

548-RQMT-0001

Release: Baseline
Effective Date: August 2009

National Aeronautics and
Space Administration

Approval: [signature]
P-3B Operations Engineer

Goddard Space Flight Center
Wallops Flight Facility
Wallops Island, Virginia

CHANGE HISTORY LOG

Effective Date	Revision	Page(s) affected	Description of Changes
Aug. 2009	Baseline		Baseline

P-3B DESIGN REQUIREMENTS

1: General Designs

The design of aircraft systems and equipment installation for use onboard the P-3 Airborne Laboratory shall follow good standard aircraft industry design practice. In addition, for aircraft and aircraft systems modifications, current FAA certification standards are to be met to the maximum extent practical, consistent with the intended mission of the aircraft. All NASA and WFF standards, requirements, and directives governing aircraft, aircraft modifications, and aircraft operations shall be met.

Acceptable and certified aircraft materials and hardware shall be used for all aircraft and aircraft systems modifications, as well as for major and significant structures related to experimenter instrumentation (i.e. racks, instrument supports, platforms, pallets, etc.). Designs shall emphasize satisfying performance requirements while minimizing weight, drag, and cost. Each design shall have an accompanying stress analysis, report, or statement, commensurate with the magnitude of criticality and complexity of the system, to show conformance with the requirements in this document. Modifications which affect existing aircraft structure or systems shall be well documented and all possible effects of changes or additions shall be considered in the design process. Revisions to affected manuals, procedural lists, inspection documents, etc. must be followed through.

Deviations, exceptions, and one-time waivers of the requirements in this document constitutes acceptance of risk above normal operations and shall be evaluated by the Airworthiness Flight Safety Review Board (AFSRB) as to determine the proper level of review required. Approvals from such reviews must be obtained prior to placing the article or system in place.

2: Mechanical Design Requirements

2.1: General Considerations

Design of aircraft structure and structures for aircraft applications shall take into account actual and worst case loading criteria such as air loads, landing loads, crash loads, ditching loads, takeoff loads, fuselage pressurization loads, gust loads, weight and inertia loads, etc. In reference to these aircraft loads, the following definitions apply:

"Limit or applied loads are the maximum loads anticipated on the airplane during its lifetime of service. The airplane structure shall be capable of supporting the limit loads without suffering detrimental permanent deformations. At all loads up to the limit loads, the deformation of the structure shall be such as not to interfere with the safe operation of the airplane."[1]

Unless otherwise specified, a factor of safety of 2 must be applied to the limit load and where applicable a fitting form factor of safety should be used and provided in the analysis. A lower factor of safety between 2 and 1.5 can be used if load testing is performed on the structure before flight. A factor of safety of less than 1.5 is not allowed.

When ultimate loads are used, a factor of safety need not be applied to the limit loads. At ultimate load, the structure may yield but must not fail.

[1] Definition extracted from Bruhn, "Analysis and Design of Flight Vehicle Structures", Jacobs Publishing, Inc., 1973.

The term margin of safety is used to evaluate the safeness of a member. The equation below shows the computations for the margin of safety (*MS*). It is determined by multiplying the design limit stress by the appropriate factor of safety and comparing it to either the yield or ultimate material allowable stress using the following equation:

$$MS = \frac{allowable\ stress}{FS\ x\ actural\ stress} - 1$$

When two or more loads (such as shear, tension, compression, bending, or torsion) act simultaneously, the margin of safety may need to be computed using interaction formulas.[2] Below is the accepted Wallops form of the interaction equation:

$$\left(\frac{(FS) * shear}{shear\ allowable}\right)^3 + \left(\frac{(FS) * \ tension}{tension\ allowable}\right)^2 = 1$$

$$MS = FS - 1$$

The allowable stress shall take into consideration factors such as stress concentrations and fatigue, as well as additional factors of safety required for particular applications where certain unknowns may warrant additional conservatism. High factors of safety are typically used in the design of pressurized components, fittings, castings, welding, windows, composite constructions, etc., and in areas where failure of a component would pose significant risk to personnel or property.

2.2: Applied Loads and Condition

Equipment and its installation must be independently strong enough to withstand the loading conditions specified below. Additional requirements for specific designs are covered in the structural design criteria for the P-3 and by the applicable CARs and FARs. The following loads and conditions are not all inclusive.

2.2.1: Cabin Pressurization [3,4,5]

The pressurization loads used in design are shown in

Table *1* In general, structures designed to withstand cabin pressurization shall be designed to ultimate pressure (2*P*) plus flight loads and aerodynamic pressure or suction effects. For designs where weight is a factor, and where a detailed analysis is provided for the modification and surrounding aircraft structure, the ultimate pressure can be reduced to 1.5*P* plus flight loads and aerodynamic pressure and hoop tension data. See Section. 2.2.10 and its references for aerodynamic loads.

An ultimate radial inward acting pressure of 1 psi shall be used for doors and window blanks. Ditching loads shall also be considered where applicable per Section 2.2.8.

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[2] Refer to MIL-HNBK-5, Rev. E, sections 1.3.3 and 1.5.3.5. Note: MIL-HNBK-5 has recently been superseded by the Metallic Materials Properties Development and Standardization (MMPDS) document, at the time of this writing the most current revision is DOT/FAA/AR-MMPDS-02.

[3] Refer to Lockheed Report 13167, "Part IV – Structural Design Loads – Fuselage", pg 4.175

[4] Refer to Federal Aviation Administration, DOT, 14CFR Chapter 1 (1-1-89 Edition), FAR 25.365.

[5] Refer to CAR 4b.

Table 1. Minimum aircraft pressure vessel design criteria.*

Design Parameter	Pressure Limit (psi)
Maximum Cabin Differential Pressure	5.66
Maximum Emergency Relief Pressure (*P*)	5.99
Design Limit Pressure (1.33*P*)	7.97
Design Ultimate Pressure (2*P*)**	11.98

* Refer to Lockheed Report 13167 "Part IV Structural Design Loads – Fuselage", pg 4.175.

** Corresponds to ground pressure test required for aircraft certification under CAR 4.b.216 and FAR 25.365.

2.2.2: Suction Pressures

Suction pressures are used in conjunction with the loads for any flight condition including the corresponding fuselage internal pressure. Lockheed Report 13167 "Part IV Structural Design Loads – Fuselage", page 4.178 (Attachment 1) presents graphically the external suction pressures for the entire fuselage surface from FS288 aft to the rear pressure bulkhead located at FS1117. These pressures are average pressures over relatively large local areas. In the case of local areas such as for the design of cut-outs, doors, hatches, etc. then the pressures presented below in Table 2 are used. These pressures are the maximum obtainable local values for each region considered and are considered limit loads.

Table 2. External pressures on fuselage cut –outs. *

Fuselage Station	Location	Delta Pressure
F.S. 218.4 – 238	Stringers 41-46	-1.67 psi
F.S. 248 – 261.18	Stringers 46 – 48	-1.42 psi
F.S. 264.82 – 288	Stringers 41 – 46	-1.22 psi
F.S. 270 – 288	Air Conditioning Access	-1.22 psi
F.S. 288 – 323	Air Conditioning Exhaust	-1.42 psi
F.S. 390 – 398	Bomb Hoist Access	-0.51 psi
F.S. 496 – 508.66	Stringers 46 – 48	-1.0 psi
F.S. 483.33 – 496	Fuel Pump Access	-1.0 psi
F.S. 553 – 570.97	Control Cable Access	-3.0 psi
F.S. 534 – 548.66	Bottom Centerline	-3.0 psi
F.S. 553 – 570.97	Bottom Centerline	-3.0 psi

* Refer to Lockheed Report 13167 "Part IV – Structural Design Loads – Fuselage", pg 4.173.

2.2.3: Vibration [6]

Equipment to be mounted anywhere on the aircraft (interior or exterior) should be capable of withstanding a 68Hz natural frequency produced by the aircraft. This frequency is produced primarily by propeller blade passage. It is most noticeable in the propeller's plane of rotation. An audible buzzing and/or beat is created in addition to a vibration that can be felt in the deck. The level of excitations at 68Hz varies widely and is dependent on several factors, namely, indicated airspeed, propeller inflow angle, propeller blade tip Mach number, and especially, propeller synchrophaser operations. Propeller blade passage pressure pulsations can induce vibrations in loose or improperly attached structural items. Sensitive items should be vibration isolated if at all possible.

2.2.4: Emergency Landing Criteria for Passenger and Crew Compartments [7,8]

The emergency landing loads used for the design of equipment located in the main cabin are shown in Table 3. The supporting structure must be designed to restrain, under all loads up to those specified, each item of mass that could injure an occupant if it came loose in a minor crash landing. Emergency landing loads act independent and do not need to be combined with other loads such as suction and pressure loads.

[6] Refer to "P-3 Aircraft Vibration" pamphlet
[7] Refer to "P-3B (N426NA) Crash and Gust Load Criteria Change" memo dated 11/23/2007
[8] Refer to Federal Aviation Administration, DOT, 14CFR Chapter 1 (1-1-89 Edition), FAR25.561.

Seats and items of mass (and their supporting structure) must not deform under any loads up to those specified above in any manner that would impede subsequent rapid evacuation of occupants. In addition, seats shall comply with all applicable FAA design standards as outlined in FAR 25.562.

Table 3. Minimum emergency landing load criteria for the main cabin.*

Load Direction	Ultimate Load Factor (g)
Forward	9.0
Down	6**
Up	2.0**
Lateral	3.0** 4.0 for Seats
Aft	1.5

* Crash load factors based on FAR Part 25.561.

** Flight load factors may exceed the emergency landing criteria requirements, see Section 2.2.9.

2.2.5: Emergency Landing Criteria for Below Cabin Floor Area

The emergency landing loads used for the design of equipment located in the below cabin floor area FS740 to FS960 are shown in Table 4.

Table 4. Minimum emergency landing load criteria for the below cabin floor area.*

Load Direction	Ultimate Load Factor (g)
Forward	3.0
Down	4.5**
Up	2.0**
Lateral	1.5**
Aft	1.5

* Refer to DC-8 Experimenter Handbook.

** Flight load factors may exceed the emergency landing criteria requirements, see Section 2.2.9.

2.2.6: Emergency Landing Criteria for Bomb Bay Area

The emergency landing loads used for the design of equipment located in the bomb bay area are shown in Table 5.

Table 5. Minimum emergency landing load criteria for the bomb bay.*

Load Direction	Ultimate Load Factor (g)
Forward	3
Down	6.5**
Up	3**
Lateral	2.5**
Aft	1.5

* Refer to "P-3B (N426NA) Crash and Gust Load Criteria Change" memo dated 11/23/2007.

** Flight load factors may exceed the emergency landing criteria requirements, see Section 2.2.9.

2.2.7: Emergency Landing Criteria for Cockpit

The emergency landing loads used for the design of equipment located in the cockpit area are shown in Table 6.

Table 6. Minimum emergency landing load criteria for the cockpit.*

Load Direction	Ultimate Load Factor (g)
Forward	20
Down	10**
Up	6**

* Refer to Lockheed Report 13167 "Part IV – Structural Design Loads – Fuselage", pg 4.212.

** Flight load factors may exceed the emergency landing criteria requirements, see Section 2.2.9.

2.2.8: Ditching Pressure [9]

Ditching pressure is variable depending on fuselage station location. The lower surface of the fuselage, and in addition the bomb bay doors, nose landing gear doors, and other structures covering cut-out areas of the fuselage structure shall be designed to the following water pressure specifications. Table 7 below shows the ditching pressure distribution. NASA currently waives all ditching load requirements for experiment installations.

[9] Refer to Lockheed Report 13167 "Part IV Structural Design Loads – Fuselage", pg 4.125

Table 7. Ditching pressure distribution.

Area Designation	Longitudinal Station in (%)	Fuselage Station	Pressure (psi) Over Lower Half of Fuselage
A	0 to 5	105 to 170	12
B	6 to 10	171 to 226	12
C	11 to 25	227 to 393	8
D	26 to 60	393 to 783	6
E	61 to 80	783 to 1004	4

2.2.9: Flight Load Factors[10]

Operational ultimate flight load factors shall be considered in addition to crash loads for the design of the aircraft systems and equipment. An additional factor of 1.33 shall be applied to seats and seat attachments. Table 8 denotes the overall gust load factors that apply to all experimenter hardware located between FS280 and FS1130. Should a design fail to meet the Flight Load Factors in Table 8 then Figures 1 and 2 may be used to customize the design for a specific flight station. However, a new analysis will be required if the installation moves to another fuselage station who's gust load requirement is not enveloped by the previous analysis. It recommended that Figures 1 and 2 be used for bomb bay installations.

Table 8. Ultimate flight load factors FS280 to FS1130.[2]

Load Direction	Ultimate Load Factor (g)
Down	10.2
Up	6.4
Side	3.2

[10] Refer to "P-3B (N426NA) Gust Load Criteria Change" memo dated 6/7/2008

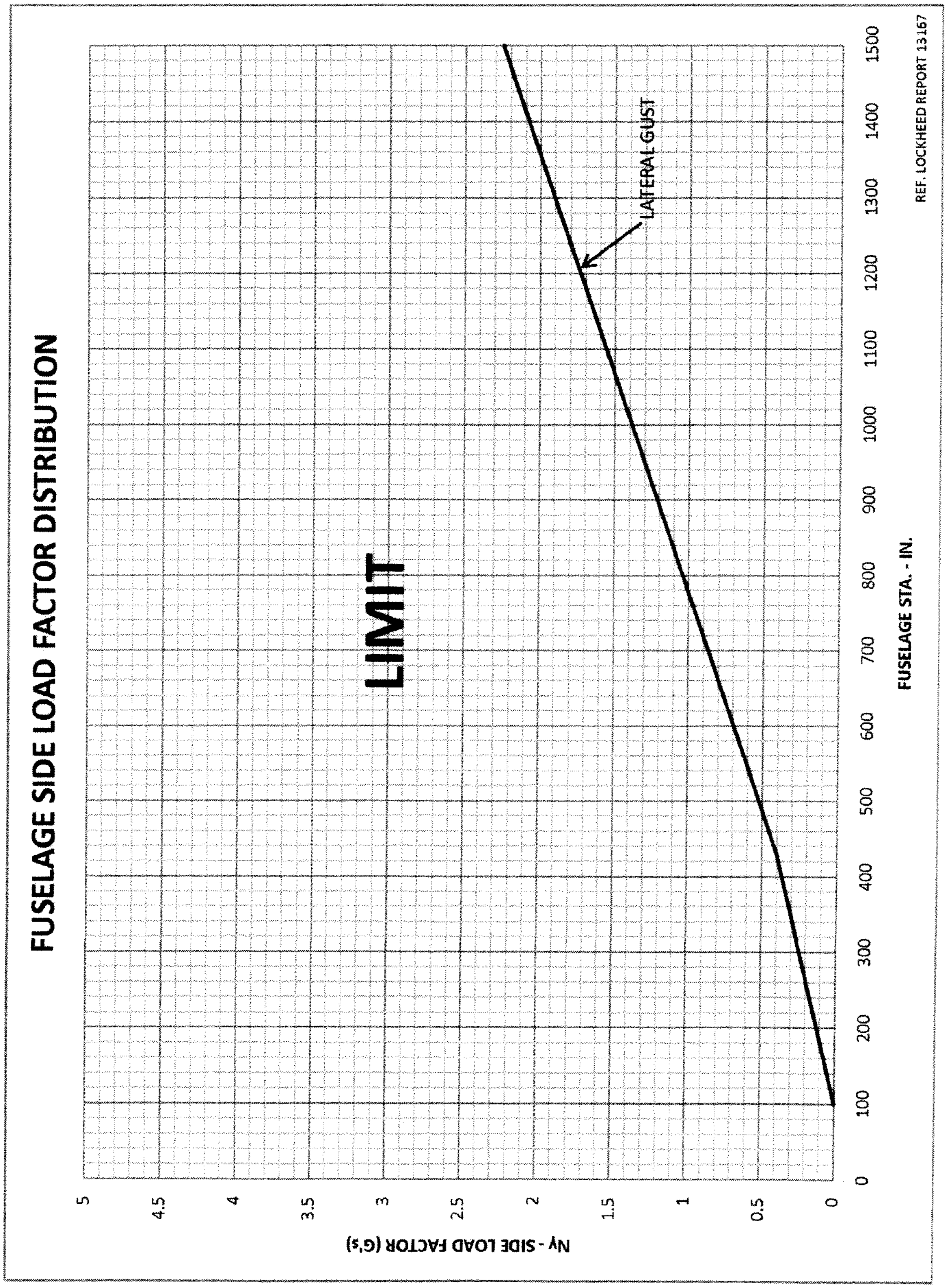

Figure 1. Lateral operational limit flight load factors.

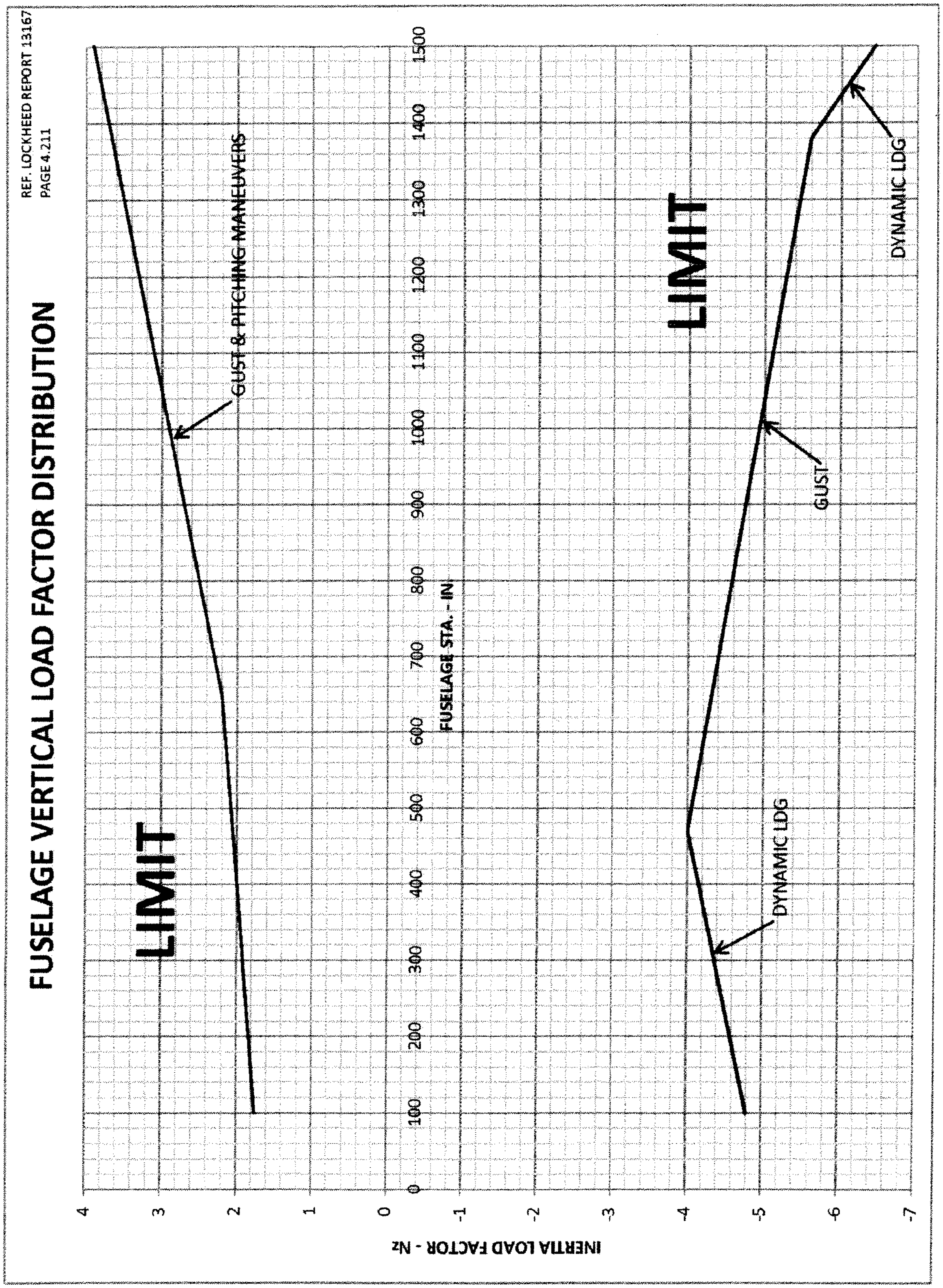

Figure 2. Downward and upward operational limit flight load factors.

2.2.10: Equipment Exterior to the Aircraft and Aerodynamic Loads

Consideration must be given to any adverse effects that equipment might have on the stability, control, and performance of the aircraft under normal and emergency conditions. The modes and impact of failures should also be considered for externally mounted equipment. Ground clearance during maximum takeoff rotation, and landing with struts compressed and tires flat should be considered for equipment mounted on the underside of the aircraft.

2.2.10.1: Ground Clearance

Installations that are hung under the aircraft and have significant height shall be checked for ground clearance issues. A 100% compressed nose strut reduces the nose landing gear by 4.75 inches. A 100% flat nose tire reduces the aircraft by 4.75 inches, thus a worse case scenario for a failed nose landing gear (compressed strut and flat tire) is a reduction of 9.5 inches of ground clearance beneath the aircraft at this location.

A 100% compressed main strut reduces the mian landing gear by 4.75 inches. A 100% flat main tire reduces the aircraft by 9.25 inches, thus a worse case scenario for a failed main landing gear (compressed strut and flat tire) is a reduction of 14 inches of ground clearance beneath the aircraft at this location.

Installations that are aft of the main gear shall be checked for ground clearance during takeoff pitch rotation. A typical takeoff pitch rotatation is approximately 5° to 7° (this takes into account the 2° nose down angle of the stationary aircraft). The average pitch angle upon landing is 5° or less.

It is recommended that any instrument installations mounted underneath the aircraft remain 4-6 inches above the ground level after the worse case ground clearance scenario is applied.

2.2.10.2: Aerodynamic Loads

The aerodynamic load of external equipment due to drag and/or lift must be calculated and put into the aircraft structure using appropriate center of pressure and moments. For this load, the dynamic pressure (q) is based on aircraft speed and atmospheric density (altitude and temperature). The dynamic pressure shall be computed at the aircraft's maximum speed, at various operating altitudes, and the greatest value for q used (3.9 psi). The absolute maximum dynamic pressure (q_{max}) for the P-3 is 555 psf (~3.9 psi). This is an ultimate load corresponding to V_d at worst conditions (sea level and 405KIAS). The maximum dynamic pressure is applied at the appropriate structure projected (planform) area. Any boundary layer reductions of loads are discarded.

Both operational load factors along with aerodynamic loads need to be considered for external structures. Conservatively "simple" aerodynamics loads can be calculated using lift and drag coefficient (C_L and C_D) values of 1.0. A limit uniform pressure coefficient of magnitude 1.0 should be applied to closeout panels in viewing apertures. The Prandtl-Glauert correction should be used to estimate the increase in loads caused by compressibility when applicable. A minimum safety factor of 2 shall be used for aerodynamic loads on exterior mounted instruments and support structures.

$$Lift = C_L Q_{max} A_{planform}(SF)$$

$$Drag = C_D Q_{max} A_{planform}(SF)$$

Reduced values of C_L and C_D can be used when more detailed calculations is appropriate. The S.F. Hoerner texts of fluid dynamics lift and drag are accepted references for coefficient values. A maximum flow incidence angle of 5-deg is appropriate for then calculating the resultant loads. Typical lift curve slope values (C_L/α) range from 0.05 to 0.08 for probe (and pylon) cross section shapes ranging from rather blunt to streamlined to airfoil. A fineness ratio (chord/thickness) of at least 3.0 is recommended to alleviate trailing vortex concerns.

2.2.10.3: Sideslip Condition [11]

Sideslip design criteria shall be applied to all externally mounted installations of substantial size. At takeoff flaps and 160 knots the maximum sideslip angle is 30.8°. This condition produces the maximum differential side pressure. An engine out scenario at altitude is also covered by this condition.

2.2.11: Ultimate Load Allowable of Seat Tracks (Cabin Floor) [12]

The floor seat tracks are designed to react to an ultimate load of 10,000 lbs. at 38" intervals anywhere along their length. The design of the floor seat track beams is to react to dynamic crash load criteria applied to all installations. Seat tracks are composed of the Boeing BAC 1520-1773 seat track material.

2.2.12: Ultimate Load Allowable of Seat Tracks (Fuselage Side Rails) [13]

The sidewall cabin mounted seat tracks are designed to react to an ultimate load of 10,000 lbs. at 38" intervals anywhere along their length. The design of the sidewall cabin seat track beams is to react to dynamic crash load criteria applied to all installations, including the standard Wallops 19" racks, if required. Side rails are composed of the Boeing BAC 1520-1773 seat track material.

2.2.13: Seat Track and Mounting Hardware Information

The following information below in Table 9 details the loading limits for the heavy duty seat track that is installed throughout the P-3 (bomb bay- Telair 20864; side walls; cabin floor; and below cabin floor – Boeing BAC 1520-1773). All seat tracks have 1 inch on center hole spacing and are made of 7075-T6 aluminum per MIL-A-8625. Single stud and double stud connectors are described below in Table 10 and are made of heat-treated, cadmium-plated alloy steel.

[11] Refer to Lockheed Report 13167 "Part IV Structural Design Loads – Fuselage", pg 4.193
[12] Refer to "N426 Supplement to Aircraft Flight Manual", pg FMS-7
[13] Refer to "N426 Supplement to Aircraft Flight Manual", pg FMS-7

Table 9. Seat track design capacity information for both Telair and Boeing tracks.*

Angle from Centerline of Track	Vertical Angle of Pull	Heavy Duty Seat Track Capacity
0°	0°	8,950 lbs
0°	30°	8,075 lbs
0°	60°	9,800 lbs
0°	90°	11,600 lbs
45°	0°	10,950 lbs
45°	30°	8,038 lbs
45°	60°	8,625 lbs
90°	0°	8,225 lbs
90°	30°	7,275 lbs
90°	60°	8,000 lbs

* Refer to Telair International 2004 Product Catalog

Table 10. Seat track stud connectors.

Load Type	Certified Single Stud Load*	Certified Double Stud Load**	Uncertified Single Stud Loads***
Tension	4,000 lbs	6,600 lbs	6,000 lbs
Shear	4,000 lbs	7,000 lbs	2,200 lbs
Bending	N/A	N/A	2,300 inch-lbs
Horizontal load along centerline and 42 inches above the track	N/A	2,750 lbs	N/A
Horizontal load perpendicular to track and 42 inches above track	N/A	3,700 lbs	N/A

* Refer to ANCRA Product Catalog, Part #'s: 40073-16 and -35

** Refer to Telair International 2004 Product Catalog, Part #: 41293-104

*** Uncertified Brownline studs, used Telair 2004 Product Catalog for single stud strengths, these studs have been marked with a "X" on the stud and retainer to defferientate from the certified single studs

Uncertified single mounting seat track studs should only be used for lightweight items such as passenger oxygen bottle and smoke mask mounting. These studs are not recommended for use on items that weigh over 100lbs and moment arms greater than 12 inches from the seat track surface.

2.2.14: Floorboards

The P-3 floor is constructed of multiple 1/2 inch thick floorboard panels called Gillfloor 5007C. These panels are sandwich panels made of polyester fiberglass facings fused to end grain balsa wood cores. Each panel is constructed of 9 pcf balsa wood cores with Gillfab 1074 mat overlays for facing material. Both materials are adhered together using a modified polyester fire resistant adhesive. Below are typical properties for the Gillfloor 5007C panels:

Table 11. Gillfloor 5007C material properties.*

Property	Test Method	Measurement
Weight	ASTM C29	1.09 lb/sq ft
Sandwich Peel	AMS-STD-401	52 in-lb/3 in width
Long Beam Flexural Ultimate Load Facing Stress Deflection @ 100 lbs. load	AMS-STD-401	 431 lb 22.0 ksi 0.629 inches
Flatwise Compression	AMS-STD-401	1,982 lb/sq in
Flatwise Tensil	AMS-STD-401	644 lb/sq in
Insert Shear, (Note: Shurlok Insert 5107-A3 used in test)	BMS 4-17	1,028 lb
Impact Strength	ASTM 3029	84 in-lb
Flammability – 60 sec. vertical Self-Extinguishing Time Burn Length Drip Extinguishing Time	FAR Part 25, Appendix F, Part I	 0 sec. 0.6 sec. 0 sec.
Flammability – 45 Degree Self Extinguishing Time Afterglow Penetration	FAR Part 25, Appendix F, Part I	 0 sec. 0 sec. None

2.2.15: DC-8 Window Allowable Loads of P-3B[14]

The design loads for the DC-8 window frame installations are:

- 1.5*P* (8.99 psi)

[14] Refer to "Engineering Report: GTE Windows FS458.5"P, 739.5"S, and 758.5"P on N4236NA" 2/20/1996

- 100 lbs (limited to 150lbs. ultimate) vertical load with C.G. 18 inches from the viewport mounting surface combined with a drag area of 1 sq. ft.
- 2.0*P* (11.98 psi) acting independently[15]

The external loads are applied to the frame by means of 32, 10-32 bolts that attach the viewport plates to the airframe.

2.2.16: Zenith Frame Allowable Loads[16]

The zenith port installation located at FS795 has a 16" diameter aperture that uses 4 aluminum clips to hold instrumentation or optical windows in place. The maximum allowable weight for objects mounted in this port is 280 lbs. with a CG 10" below the port opening.

2.2.17: P-3B Bubble Window Allowable Loads[17]

The design loads for the P-3 bubble window frame installations are similar to the DC-8 window allowables:

- 1.5*P* (8.99 psi)
- 100 lbs (limited to 150lbs. ultimate) vertical load with C.G. 18 inches from the viewport mounting surface combined with a drag area of 1 sq. ft.
- 2.0*P* (11.98 psi) acting independently[2]

The external loads are applied to the frame by means of 36, ¼-28 bolts that attach the viewport plates to the airframe.

2.2.18: Optical Window Design

All optical windows (excluding standard P-3 aircraft windows) shall comply with the window design standards as outlined in P-3B "Design Elements for Aircraft Optical Windows, 548-RQMT-0002, July 2009". All optical windows shall comply with the "P-3B Optical Window Inspection Plan, 548-PLAN-0001, July 2009". Copies of both documents can be obtained from the ops engineer.

2.2.19: Loads for Tables and Workstations

A limit load of 250 pounds (ultimate load of 500 pounds) shall be used for loads on tables, workstations, or any structure where there is a probability for someone to stand, sit, or otherwise support their weight on the structure.

2.3: Load Test

Items requiring load test must comply with "NASA-STD-5001, Structural Design and Test Factors of Safety for Spaceflight Hardware." Contact the ops engineer for a current version of this NASA technical standard.

2.4: Additional Design Constraints and Guidelines

The following are general design constraints and guidelines to use in the design of hardware for use on the P-3.

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[15] Refer to CAR 4b.216.
[16] Refer to 548-STR-0112 "P-3B Orion (N426NA) 16" Zenith Port Loading Envelope Analysis" 4/2008
[17] Refer to 548-STR-0155 "P-3B Orion (N426NA) P-3 Bubble Window Stress Analysis" 2/2009

1. Welded structures should be avoided for significant load bearing and/or externally mounted structure. Where necessary, use stainless steel or steel with appropriate corrosion resistance measures taken.
2. Nutplates through fuselage skins and other significant load bearing structures shall be avoided; gang channels and secondary plates with nutplates are acceptable alternatives.
3. Open holes through the fuselage greater than 0.375 in diameter shall be reinforced with a skin doubler. The edges of the holes should be reamed to a 63 micro-in finish.
4. Maintain minimum fastener edge distance spacing of twice the fastener diameter (2*d*).
5. Maintain minimum fastener to fastener spacing of four times the fastener diameter (4*d*).
6. Avoid open holes and nutplates through frame flanges (caps) and other significant load bearing structures.
7. Allow for drainage where there is a likelihood of water entrapment.
8. Provide breathing holes in enclosed structures, especially outside the aircraft, to prevent over-pressurization loads at altitude.
9. Provide for proper corrosion prevention and control[18]
10. Avoid threaded inserts into significant load bearing structure (i.e. skin, window blanks, frame caps, etc.)
11. All designs involving the flight deck, aircraft exterior or significant weight/balance changes must be cleared and signed off by the appropriate review board.
12. Special windows used by experimenters are considered primary structure. However, they have unique and more restrictive design requirements including inspection.[19]

3: Safety

All practical and necessary steps will be taken to avoid the loss of life, personal injury, property loss, mission failure, or test failure. All potential hazards associated with the operation of new or modified systems must be identified and the risk associated with each identified hazard must be assessed and reduced, if necessary, to an acceptable level.

The failure of a single element (hardware or software) or the commission of a single operator error having a remote or higher probability of occurrence must not cause a critical or catastrophic hazard.

Potential hazards shall be identified, as required, through some (or all) of the processes listed below, as necessary and applicable.

- Configuration Reviews
- Preliminary Design Reviews
- Critical Design Reviews
- Final Installation and Inspection Reviews
- Hazard and Failure Analysis

[18] Refer to P-3 Structural Repair Manual NAV 01-75PAA-3-1.1
[19] Refer to 548-PLAN-0001 "P-3B Orion (N426NA) P-3B Optical Window Inspection Plan", 7/ 2009

- Flight Readiness Reviews
- Mission Readiness Reviews

Potential hazards should be resolved in the following order:

1. Eliminate the potential hazard through redesign.
2. Control the potential hazard by the incorporation of safety devices.
3. Control the potential hazard by the incorporation of alarming devices.
4. Control the potential hazard by operation procedure(s).

All installations are subject to airworthiness and flight safety review, as necessary and applicable.

4: Definitions

4.1: Primary Structure

Primary structure is structure, which upon failure will cause major loads to be transferred to the surrounding structure. These major loads may result in progressive failure of the surrounding structure and possibly the loss of the airplane.

4.2: Secondary Structure

Secondary structure is structure, which can fail and have the surrounding structure successfully carry the loads formerly carried by the failed structure. Although the damaged structure should be repaired as soon as possible, it will not present an immediate hazard.

4.3: Limit (Applied) Loads

"Limit or applied loads are the maximum load anticipated on the airplane during its lifetime of service. The airplane structure shall be capable of supporting the limit loads without suffering detrimental permanent deformations. At all loads up to the limit loads, the deformation of the structure shall be such as not to interfere with the safe operation of the airplane."[20]

4.4: Ultimate Loads

Ultimate loads are the maximum loads for which the structure may yield but must not fail. The ultimate loads are typically defined as the limit loads times a factor of safety. A factor of safety of 2 is used for equipment designed for use on the P-3.

4.5: Load Factor

Load factor is a multiplying factor which defines a load in terms of weight. It is a measure of the magnitude of additional loading on the aircraft structure due to aircraft maneuvers, flying through turbulence, etc.

4.6: Airload

Airloads are loads resulting from aerodynamic forces that are applied to the surfaces of a given structure.

[20] Definition extracted from Bruhn, "Analysis and Design of Flight Vehicle Structures", Jacobs Publishing, Inc., 1973.

5: Basic Checklist for Preparing Engineering Data for a Design

Limit Loads
Flight Loads
Cabin Limit Design Pressure
Thermal Loads/Stresses
Aerodynamic Load Factor
Aerodynamic Loads (Loads time Load Factor)
Stresses from Limit Loads – Must be less than the material yield strength
Limit Load Margins of Safety – Must be positive

Ultimate Loads – Must be more than 2 times the Limit Loads
Design Ultimate Cabin Pressure – Do not add this to crash loads
Passenger Area Crash Loads
Below Cabin Floor Crash Loads
Bomb Bay Crash Loads
Ditching Pressure
Stresses from Ultimate Loads – Must be less than the material ultimate strength
Ultimate Load Margins of Safety – Must be positive

Misc. Design Data
P-3B Bubble Window Frame Allowable Loadings
DC-8 Viewport Frame Allowable Loadings
Zenith Viewport Frame Allowable Loadings